NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC APOLLO 13 INVESTIGATION TEAM

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PANEL 5A

CORRECTIVE ACTION STUDY AND IMPLEMENTATION

FOR

COMMAND AND SERVICE MODULES

JUNE 1970

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

PANEL 5A REPORT

MSC APOLLO 13 INVESTIGATION TEAM

PANEL 5A, COMMAND AND SERVICE MODULE (CSM)

CORRECTIVE ACTION STUDY AND IMPLEMENTATION

June 12, 1970

APPROVED BY: A. C. Cohen,
Panel Leader

MSC APOLLO 13 INVESTIGATION TEAM

CORRECTIVE ACTION STUDY AND IMPLEMENTATION

COMMAND AND SERVICE MODULE

PANEL 5A

Panel Leader - Aaron C. Cohen

Team Members - R. Ferguson (EP) Subsystem Analysis and Design

C. Dumis (FC) Flight Operations

J. Cooper (PT) Ground Support Equipment

W. Hoyler (PT) Subsystem Analysis and Design

L. Spence (NA) Reliability and Quality Assurance

C. Rice (SN) Flight Safety

R. Bobola (PF) Configuration Assessment

D. Teegarden (PF) Planning and Reporting

R. Hood (PP) Procurement and Contracts

G. Merrick (North American Rockwell) Contractor Coordination

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1.0 SCOPE

This document presents the report of the CSM Corrective Action Study and Implementation Panel (Panel 5A) of the Apollo 13 Investigation Team. The material presented herein covers the three areas of activity which were the responsibility of Panel 5A:

1. Study and implementation of corrective action associated with the cryogenic oxygen system failure which occurred on Apollo 13.

2. Establishment of the as-flown configuration of SM 109 Bay IV and identification of differences within the bay between SM 108, SM 109, and SM 110.

3. Assessment of the feasibility and implementation of corrective action for potential problems identified by the Related Systems Evaluation Panel (Panel 6).

2.0 INTRODUCTION

Panel 5A activities were initiated immediately after completion of the Apollo 13 flight. The initial effort was aligned towards alternate design approaches which would circumvent the loss of the No. 2 oxygen tank as well as the secondary anomaly relative to the inadvertent closure of two of the three fuel cell oxygen shut-off valves. By April 19, 1970, the Spacecraft Incident Investigation Panel (Panel 1) had identified the cause for the loss of tank No. 2 as combustion which resulted in overpressurizing and/or overheating of the tank and related components. Design features of existing tank internal components are shown for reference in Figure 1.

A second issue under the purview of Panel 5A was the establishment of the configuration of the hardware in bay IV of the service module SN 109, and the assessment of the significance of any differences among service modules SN's 108 (Apollo 12), 109 (Apollo 13), and 110 (Apollo 14). This review effort was performed by elements of MSC and the spacecraft contractor, North American Rockwell (NR).

Subsequently, a third issue was added to the scope of this panel when Panel 6 identified a potential problem with the oxygen fuel cell shut-off valves. The coils and associated teflon coated wiring are exposed to the supply oxygen which is normally at 900 psia and essentially at ambient temperature.

Due to the fact that Panel 5A was operating in parallel with Panel 1, the initial design approach for corrective action had to be broad to insure encompassing the final determinations of the investigation. This approach was updated and modified as warranted.

① QUANTITY GAUGING PROBE

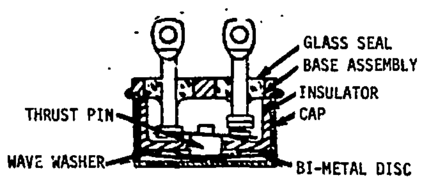
THE QUANTITY GAUGING PROBE CONSISTS OF TWO CONCENTRIC TUBES WHICH SERVE AS CAPACITOR PLATES, WITH THE OPERATING MEDIA ACTING AS THE DIELECTRIC BETWEEN THE TWO. THE DENSITY OF THE FLUID IS DIRECTLY PROPORTIONAL TO THE DIELECTRIC CONSTANT AND THEREFORE PROBE CAPACITANCE.

RANGE	0-100% FULL (1.39-69.5 LB/FT^3)
ACCURACY	±2.68% FULL RANGE
OUTPUT VOLTAGE	0-5 V DC
OUTPUT IMPEDANCE	500 OHMS
POWER	2-1/2 WATTS 115 V 400 CPS

② TEMPERATURE SENSOR

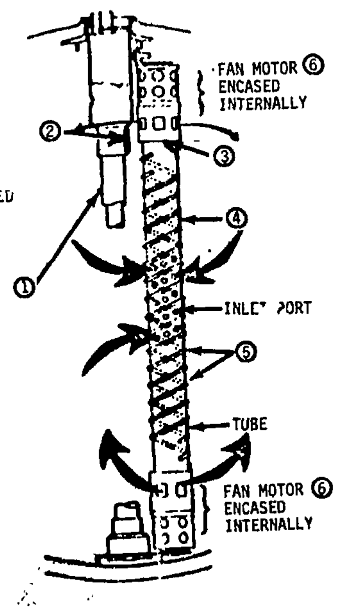
THE TEMPERATURE SENSOR IS A FOUR-WIRE PLATINUM RESISTANCE SENSING ELEMENT MOUNTED ON THE QUANTITY GAUGING PROBE. IT IS A SINGLE POINT SENSOR ENCASED IN AN INCONEL SHEATH WHICH ONLY DISSIPATES 1.5 MILLIVOLTS OF POWER PER SQUARE INCH TO MINIMIZE SELF-HEATING ERRORS.

RANGE	-320°F TO +80°F
ACCURACY	±2.63% FULL RANGE
OUTPUT VOLTAGE	0-5 V DC
OUTPUT IMPEDANCE	5000 OHMS
POWER	1.25 WATTS 115 V 400 CPS



③ THERMOSTAT

THE THERMOSTATS ARE A BIMETAL TYPE UNIT DEVELOPED FOR CRYOGENIC SERVICE. THEY ARE IN SERIES WITH THE HEATERS AND MOUNTED INSIDE THE HEATER TUBE WITH A HIGH CONDUCTING MOUNTING BRACKET ARRANGED SO THAT THE TERMINALS PROTRUDE THROUGH THE TUBE WALL. WHEN THE THERMOSTATS REACH 80 ± 10° F, THE THERMOSTATS OPEN CUTTING POWER TO THE HEATERS TO PREVENT OVER HEATING OF THE PRESSURE VESSEL. WHEN THE TUBE REACHES -75°F IN THE OXYGEN TANK THE THERMOSTATS CLOSE ALLOWING POWER TO BE SUPPLIED TO THE HEATERS.



④ PRESSURIZATION AND DESTRATIFICATION UNIT

EACH OF THE STORAGE TANKS CONTAINS A FORCED CONVECTION PRESSURIZATION AND DESTRATIFICATION UNIT. EACH UNIT CONSISTS OF THE FOLLOWING:

a. A 2.0 INCH DIAMETER SUPPORT TUBE APPROXIMATELY 3/4 THE TANK DIAMETER IN LENGTH.

b. TWO HEATERS.

c. TWO FAN MOTORS.

d. TWO THERMOSTATS.

THE TUBE PROVIDES A LARGE SURFACE AREA FOR EFFICIENT HEAT TRANSFER, AND IS SMALL ENOUGH TO BE INSTALLED THROUGH THE PRESSURE VESSEL NECK. THE HEATERS ARE PLACED ALONG THE TUBE'S OUTER SURFACE AND BRAZED IN PLACE. A MOTOR-FAN IS MOUNTED AT THE UPPER AND LOWER ENDS OF THE TUBE, WHICH DRAW FLUID THROUGH THE INLET PORTS LOCATED ALONG THE TUBE, FORCE IT ACROSS THE HEAT TRANSFER SURFACE AND EXPEL IT NEAR THE TOP AND BOTTOM OF THE VESSEL.
BLOCK II TANKS UTILIZE SEPARATE SETS OF LEAD WIRES FOR EACH HEATER ELEMENT AND FOR EACH MOTOR FAN THROUGH THE ELECTRICAL CONNECTOR INTERFACE.

⑤ HEATERS

THE HEATERS ARE A NICHROME RESISTANCE TYPE, EACH CONTAINED IN A THIN STAINLESS STEEL TUBE INSULATED WITH POWDERED MAGNESIUM OXIDE. THE HEATERS ARE DESIGNED FOR OPERATION AT 28 VOLTS DC DURING IN-FLIGHT OPERATION, OR 65 VOLTS DC FOR GSE OPERATION TO PROVIDE PRESSURIZATION WITHIN THE SPECIFIED TIME. THE HEATERS ARE SPIRALLED AND BRAZED ALONG THE OUTER SURFACE OF THE TUBE. THE HEATERS ARE WIRED IN PARALLEL TO PROVIDE HEATER REDUNDANCY AT HALF POWER. THE HEATERS HAVE SMALL RESISTANCE VARIATION OVER A TEMPERATURE RANGE OF +80°F TO -420°F. HEATER POWER IS 114 WATTS BASED ON 28 VDC AT CM MAIN BUSES.

⑥ FAN MOTORS

THE MOTORS ARE THREE PHASE, FOUR WIRE, 200 VOLTS A.C. LINE TO LINE, 400 CYCLES MINIATURE INDUCTION TYPE WITH A CENTRIFUGAL FLOW IMPELLER. THE MINIMUM IMPELLER SPEED OF THE OXYGEN UNIT IN FLUID IS 1800 RPM WITH A TORQUE OF 0.90 IN. OZ. TWO FANS AND MOTORS ARE USED IN EACH VESSEL.

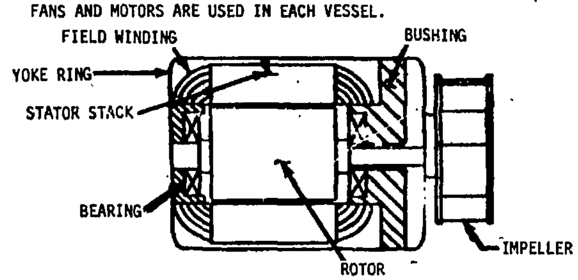


FIGURE 1. EXISTING TANK INTERNAL COMPONENTS

3.0 CONCLUSIONS AND RECOMMENDATIONS

3.1 CONCLUSIONS

It was concluded by Panel 5A that:

1. Either a capacitance probe for direct quantity measurement must be retained or additional oxygen must be provided.

2. Retention of a fan may be required based on a potential high flow requirement at low quantity levels.

3. The cryogenic oxygen tank heater control system should be revised to enhance redundancy.

4. A sensor is required to monitor the heater assembly temperature during flight.

5. The following design options are worthy of further investigation.

   a. A modified existing configuration which allows encasement of electrical leads in metal sheaths near the pressure vessel wall. This design also requires modification of the tank closeout plug to eliminate the need for turning the quantity gauging probe during assembly.

   b. A single probe incorporating the gauging probe and heaters on a tandem design.

   c. A configuration utilizing an internal gauge, an external heater and an external pump.

   d. A configuration utilizing an internal gauge, an internal heater, and an external pump.

   e. A configuration utilizing an internal gauge and an internal radiation device.

   f. A configuration utilizing an internal gauge, an internal heater, and an external gaseous oxygen pressure system for destratification.

   g. A configuration utilizing an internal gauge, an internal heater, and an internal bimetallic fan.

   h. A configuration utilizing an internal gauge, an internal heater, and vehicle movement for destratification.

6. Inadvertent closure of the fuel cell reactant valve was a resultant effect of the oxygen tank failure and was probably caused by structural shock. Thus, a feature is required in the caution and warning system to alert the crew to inadvertent closure of individual fuel cell reactant valves.

7. There were no significant configuration differences in bay IV hardware of service modules SN's 108, 109, and 110.

8. The existence of uninspectable Teflon-covered wiring next to the pressure vessel and Kel-F exposed to high pressure oxygen within the fuel cell reactant valve poses a potential failure point, and resolution of this point is required.

3.2 RECOMMENDATIONS

Panel 5A recommends that:

1. If redesign to correct the cause of the Apollo 13 failure is required, the design option to be incorporated should be based upon modifying the existing configuration. A design with metal sheathed wires near the pressure vessel wall and an improved assembly technique is feasible. This approach is most suitable since it preserves the configuration which has a significant background of flight experience. It also meets the criterion of lowering the ignition and fuel sources within the oxygen tank to an acceptable level. This design must be demonstrated safe for flight by test.

2. If destratification fans are eliminated, changes should be implemented to the spacecraft wiring to enhance redundancy.

3. Changes should be implemented to eliminate the potential problem associated with oxygen exposure of uninspectable Teflon-covered wires near the pressure vessel wall and Kel-F within the fuel cell reactant valves.

4. Changes should be implemented to the spacecraft wiring for the caution and warning system so as to provide an alert to the crew if one fuel cell reactant valve inadvertently closes.

4.0 DISCUSSION

4.1 CRYOGENIC OXYGEN TANK NO. 2 FAILURE

The initial effort of Panel 5A was to establish an evaluation of specific alternate design approaches that would eliminate all ignition and fuel sources. However, the evaluation was modified to pursue the achievement of three design goals.

1. Eliminate exposed ignition sources within the cryogenic oxygen tanks.

2. Reduce to an acceptable level the sources of fuel for combustion thin within the cryogenic oxygen tanks.

3. Maintain existing configuration so as to take advantage of previous flight experiences.

Initially, three options were evaluated by the panel.

Option 1 - Disconnect the fan-motors and quantity probe. This required no modification to the cryogenic dewar but did not meet design goals 1 and 2.

Option 2 - Encase the fan, quantity probe, and heater leads in metal sheaths. This satisfies the design goals, if adequately proven by test.

Option 3 - Disconnect fan-motors, quantity probe and heaters. This satisfied only goal 1. This approach does require the use of an external fan-motor or pump.

North American Rockwell concluded that the internal portions of existing oxygen tanks could be modified. Based on this information and the fact that the options not requiring tank refurbishment, Options 1 and 3, did not adequately meet the established criteria, they were dropped from further immediate consideration.

The primary concern with options 1 and 3 was the elimination of the capacitance type quantity gauging probe which, in spite of extended periods in flight without fan-motor operation, has proved to be accurate to within 2 or 3 percent. Alternate methods for determining quantities were evaluated and are discussed in Appendices A and B. No suitable method was identified that would not impose an additional redline penalty of approximately 75 pounds per tank. The resultant conclusion is that either the capacitance probe must be retained or additional oxygen must be provided.

A second concern with options 1 and 3 previously outlined is the potential thermal stratification of the fluid as a result of heater operation in a low gravity environment. Normally the fans are cycled twice a day, just before and just after the rest period; however, on Apollo 8 the fans were not cycled for a period of approximately 56 hours during transearth coast. The analysis of stratification based on the flight data is included in Appendix C and two significant points are summarized here.

PRESSURE DECAY EFFECTS. - This effect is only possible where the fluid properties differ significantly from an ideal gas. For 900 psia oxygen, the fluid properties rapidly achieve ideal gas characteristics after the temperature rises above the critical temperature. The fluid temperature does not attain this range until the quantity remaining is less than 30 percent. The greatest decays in pressure have been noted with essentially full tanks and are not present below 60 percent quantity remaining. The most probable explanation is that the amount of heat required to maintain a specific mass flow from the tank is drastically reduced in the mid-quantity range, and heater off periods are normally several hours.

A special area of concern is potential residual stratification at launch resulting from heating the oxygen from an initial subcritical or two-phase condition. Adequate techniques can most likely be devised for assuring a fairly uniform condition at launch, and a test program is under way in the MSC Thermochemical Test Area to determine if a problem exists based on the present launch timelines.

HEATER SURFACE TEMPERATURE. - A second factor to consider relative to stratification is the possibility of overtemperature of the electrical heaters. All tanks flown to data have been equipped with thermal switches to cut off heater power when the thermostat reaches +80°F, which is the maximum allowable tank wall temperature. In no case has an incident of power interruption as a result of normal thermal switch action been noted in flight. However, no flight has required extended oxygen flows corresponding to those required for EVA, and more significantly, no flight experience has been gained with oxygen quantities below 30 percent full. While thermal stratification would not cause pressure decays in this region, they are expected and will tend to be increasing in significance with decreasing quantities. In addition, the bulk temperature of the oxygen at 5.5 percent remaining reaches +80°F irrespective of stratification.

It is not known if heater overtemperature conditions can be overcome with operational procedures coupled with a potential minor adjustment of the mission redlines. For this reason, complete deletion of some means of circulating this fluid past the heater cannot be justified at this time. To date, retention of an internal fan appears to be the best solution to this problem.

As a result of the stratification analysis several steps are being pursued in parallel.

1. Both NR and MSC will undertake an effort to establish a reasonable heat transfer analysis model to better understand thermal stratification under actual flight conditions.

2. A temperature sensor is planned for any heater configuration considered. This information is essential in the high flow and/or low density regions since the thermal switch had previously been deleted effective on CSM 114. Heater temperature data can be used for verifying the analysis model in the normal density and flow regimes. With verification in these regions, the analytical model can be used with better confidence in the extreme regions.

3. The MSC Structures and Mechanics Division (SMD) has been requested to: (1) establish the maximum allowable temperature gradient to which the pressure vessel may be subjected, (2) reevaluate the temperature limitation on the pressure vessel, (3) establish an upper temperature limit on the heater itself. The heater temperature will be limited to the lowest value as determined by these three approaches.

4. Because of the uncertainty due to the pressure decay effects and heater surface temperature, MSC is performing additional conceptual designs for the following:

a. A configuration utilizing an internal gauge, an external heater, and an external pump.

b. A configuration utilizing an internal gauge and an internal radiation device.

c. A configuration utilizing an internal gauge, an internal heater, and an external gaseous oxygen pressure system.

d. A configuration utilizing an internal gauge, an internal heater, and an internal bimetallic fan.

e. A configuration utilizing an internal gauge, an internal heater, and vehicle movement for destratification.

Of the initially considered options, with only that of encasing the fan, quantity probe, and heater leads in metal sheaths (option 2) remains in primary consideration. The variations in this option (Figure 2) which remain under study are:

Option 2A. - Side-by-side heaters/probe assembly and bottom fan with leads encased in metal sheaths. This requires modification of the tank closeout in order to eliminate the need for turning the quantity gauging probe (Figure 3).

Option 2B. - Side-by-side heaters/probe assembly with leads encased in metal sheath. This requires the same closeout modification as 2A.

Option 2C. - Single assembly with tandem heater/probe.

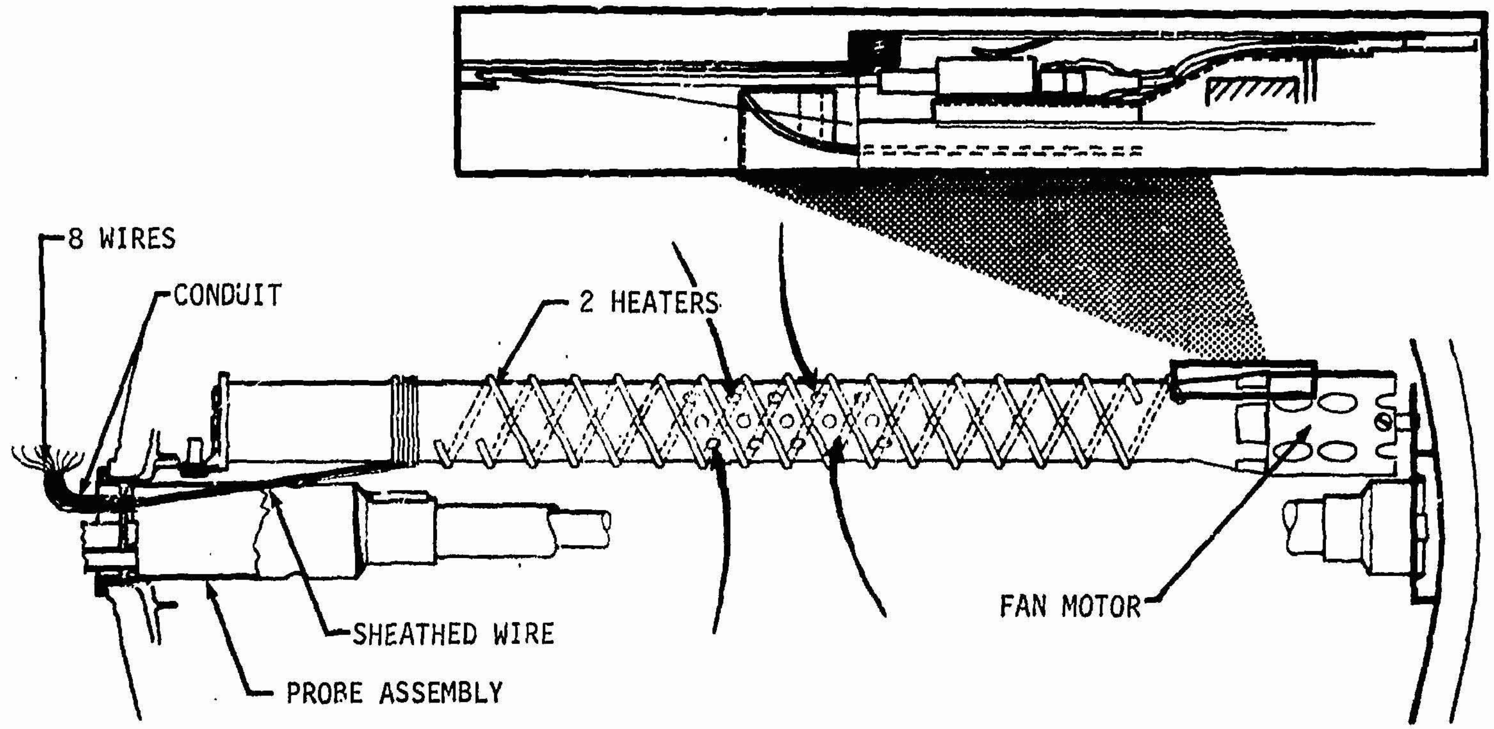


FIGURE 2. (OPTION 2A WITH FAN) QUANITY PROBE/HEATER DESIGN OPTION.

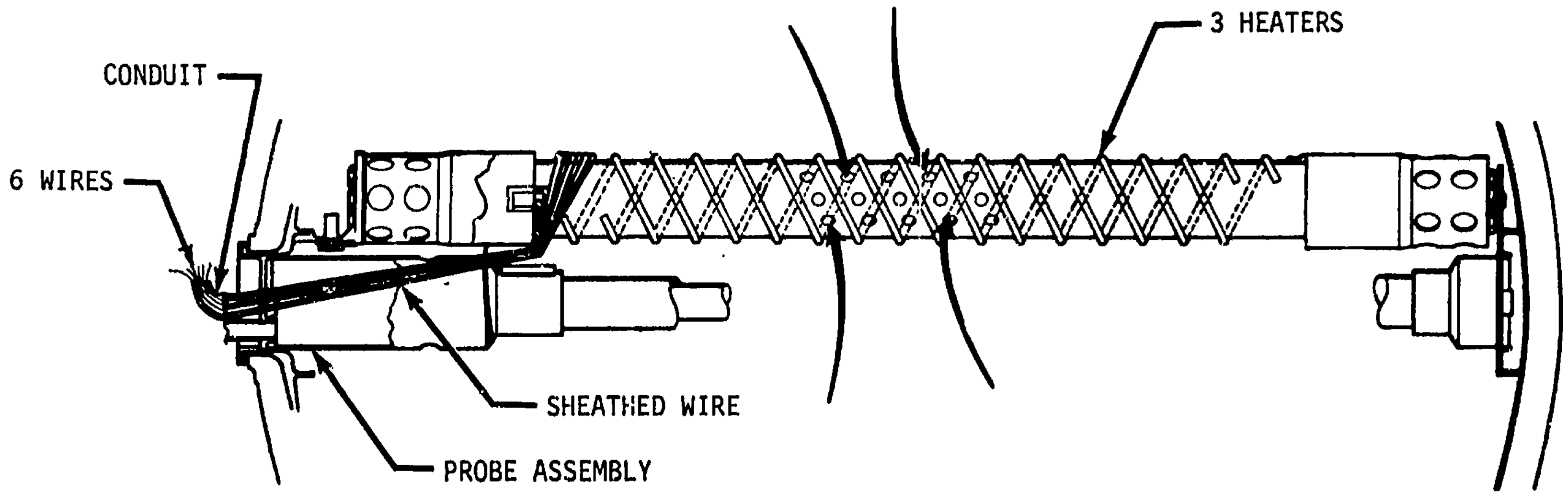


FIGURE 2. (OPTION 2B WITHOUT FAN) QUANITY PROBE/HEATER DESIGN OPTION.

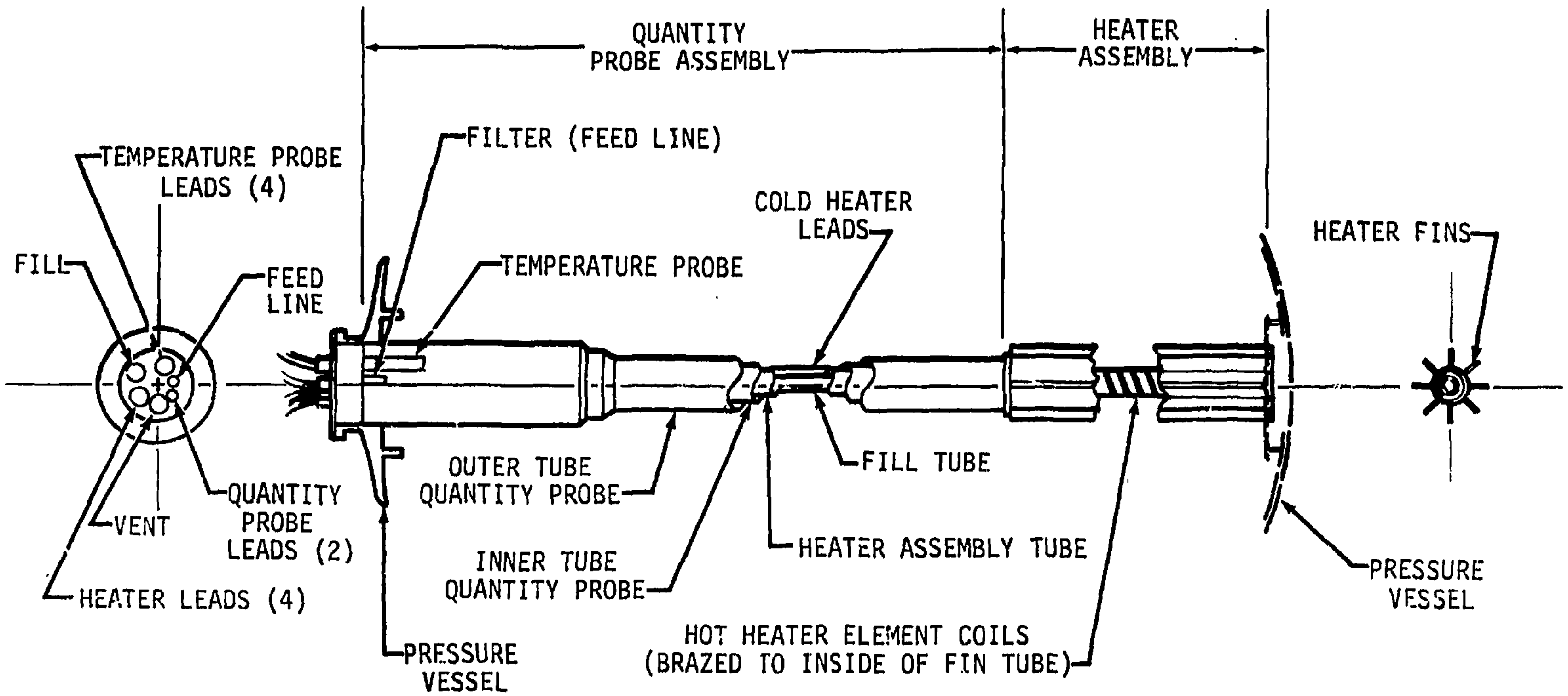


FIGURE 2. (OPTION 2C TANDEM CONFIGURATION) QUANITY PROBE/HEATER DESIGN OPTION.

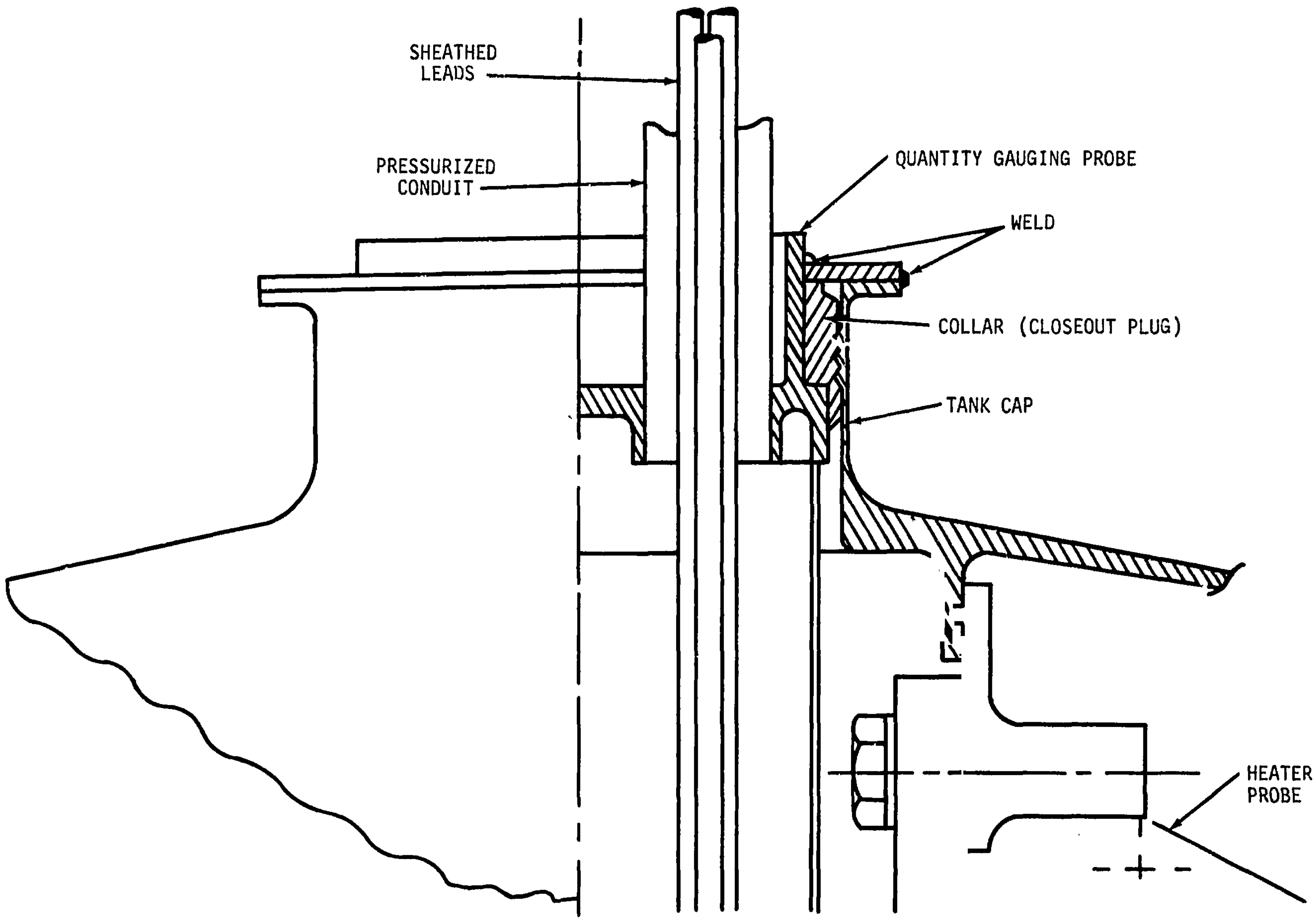


FIGURE 3. OPTIONS 2A AND 2B. MODIFIED TANK CLOSEOUT PLUG.

Option 2C meets the criterion and overcomes the fabrication concerns; however, any configuration other than the side-by-side compromises the usefulness of previous flight data.

As the design efforts have matured, several specific design issues related to these options have been identified. These areas which require further analysis and/or development before implementation are discussed in the following paragraph.

1. What is the heater capacity requirement? - NR has determined that the available heater lead wire at the heater vendor's plant is not suitable for carrying more than 1.7 amps. As a result, NR has proposed that the oxygen tank be equipped with three heater elements with two elements in parallel on one of the two independent heater circuits. Since the third heater and associated leads through the annulus and tank penetration constitutes added complexity, the panel undertook a study to reevaluate the flow requirements for all missions.

With the calculated system resistance exclusive of the heaters of 1.7 ohms, the 1.7-ampere limitation results in a net of 42.5 watts per heater. An analysis was made using one, two, and three heaters per tank, as presented in Appendix D. The results of this analysis indicate that a two heater configuration will satisfy all normal and contingency requirements as well as the three heater configuration; however, there are some extreme cases in which the pressure will decay during short term, high flow rate periods. In all cases these were deemed acceptable from a system operational standpoint.

At the low density or low quantity region the limited heater surface area will override the amount of energy dissipation required. Therefore, the third heater will not be effective in this region. As a result, it is not expected that the detailed low gravity heat transfer analysis mentioned previously will show any advantage for the third heater in the low density region.

The two heater configuration is a reasonable option if the installation of the third heater element constitutes a major design problem.

2. Which configuration will provide the best combination of gauging accuracy and maintenance of pressure? - Tests will be performed at Beech Aircraft Company to determine heater performance and gauging accuracy differences between the configurations of Options 2b and 2c.

3. What ground servicing procedures are required to prevent severe stratification during ground operations? - Procedures will be developed through testing in the MSC Thermochemical Test Area (Reference Appendix E).

4. What is the possibility of excessive heater temperature due to stratification and elimination of the heater thermoswitch? - North American Rockwell (NR) will provide a heater temperature sensor and MSC will establish the temperature gradient limits for the oxygen tank material to provide a reference for defining probe operating temperature limits.

5. What are the heater fin configuration requirements for the tandem probe design? - NR will evaluate the use of radial versus vertical fins.

6. What is the effect on insulation resistance of the heater element and sheathed leads caused by the small radius bends around the heater body and through the conduit from the tank cap to the external connector? - This issue will be evaluated through testing by the heater vendor.

7. Is the glass impregnated teflon insulator material used in the quantity gauging probe assembly suitable for retention in the final design? - MSC will complete an evaluation including strength of the material at cryogenic temperatures, oxygen compatibility, and potential for igniting small chips of the material versus potential for igniting quantities represented by the actual probe components. Pending resolution of this issue, NR was directed to proceed with design using the teflon material.

8. Can the incorporation and use of the fan in the cryogenic oxygen system be accomplished safely? A detailed test and evaluation program has been initiated to answer this question.

Design features for either probe configuration are the following:

1. Use of Inconel instead of aluminum in the body of the quantity gauging probe to eliminate a potential fuel source. This feature was verified as an acceptable step by the probe vendor.

2. Use of two independent heater circuits wired and controlled as shown in figure 4 to replace the redundancy lost if the fans are eliminated.

3. Use of a pressurized conduit to route the electrical leads from the tank cap to the external connector (see figure 5).

4.2 INADVERTENT FUEL CELL REACTANT VALVE CLOSURE.

One of the associated effects of the incident to the spacecraft was the closing of two of the three fuel cell reactant valves in the fuel cell valve module. It was determined that the reactant valve closure was a resultant effect of the oxygen tank failure and was probably caused by structural shock. This effect of inadvertent valve closure is significant since a decay of either reactant regulated pressure of eight psi (pounds per square inch) or more in the stack causes failure of the fuel cell.

The fuel cell reactant supply valves are operated by solenoids with mechanical latching mechanisms. The present spacecraft installation includes a holding circuit on the battery relay bus which is energized during high 'g' periods, such as launch vibration, to prevent inadvertent valve closure. Also, the crew 'talkback' for valve positions are wired with hydrogen and oxygen valves in series such that both reactant supply valves to a specific fuel cell must close in order for the crew to get a 'barberpole' indication. This deficiency could be eliminated by a minor wiring change which would give the crew a 'barberpole' if either valve closed.

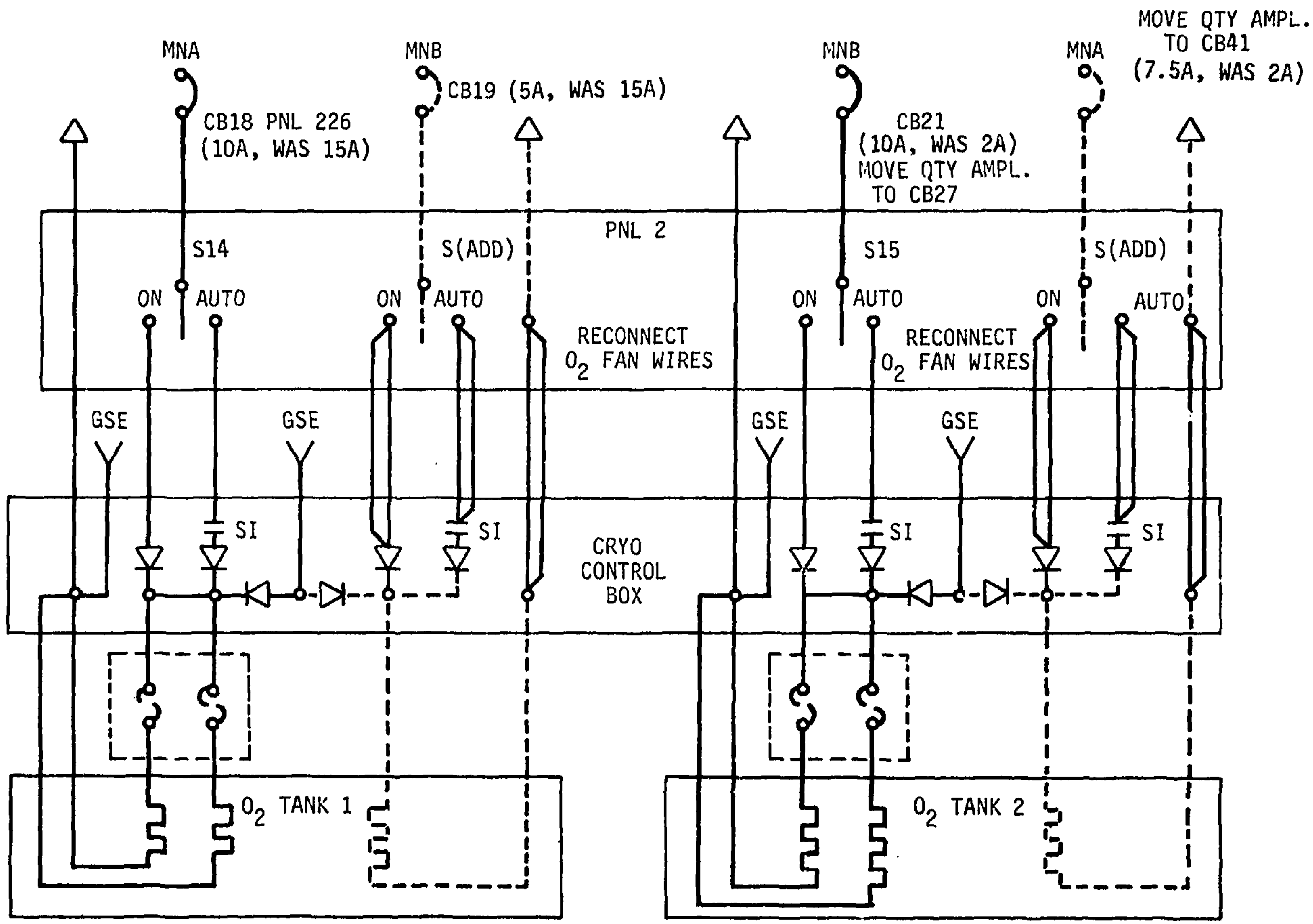


FIGURE 4. TWO OXYGEN TANKS, THREE HEATER CONFIGURATION.

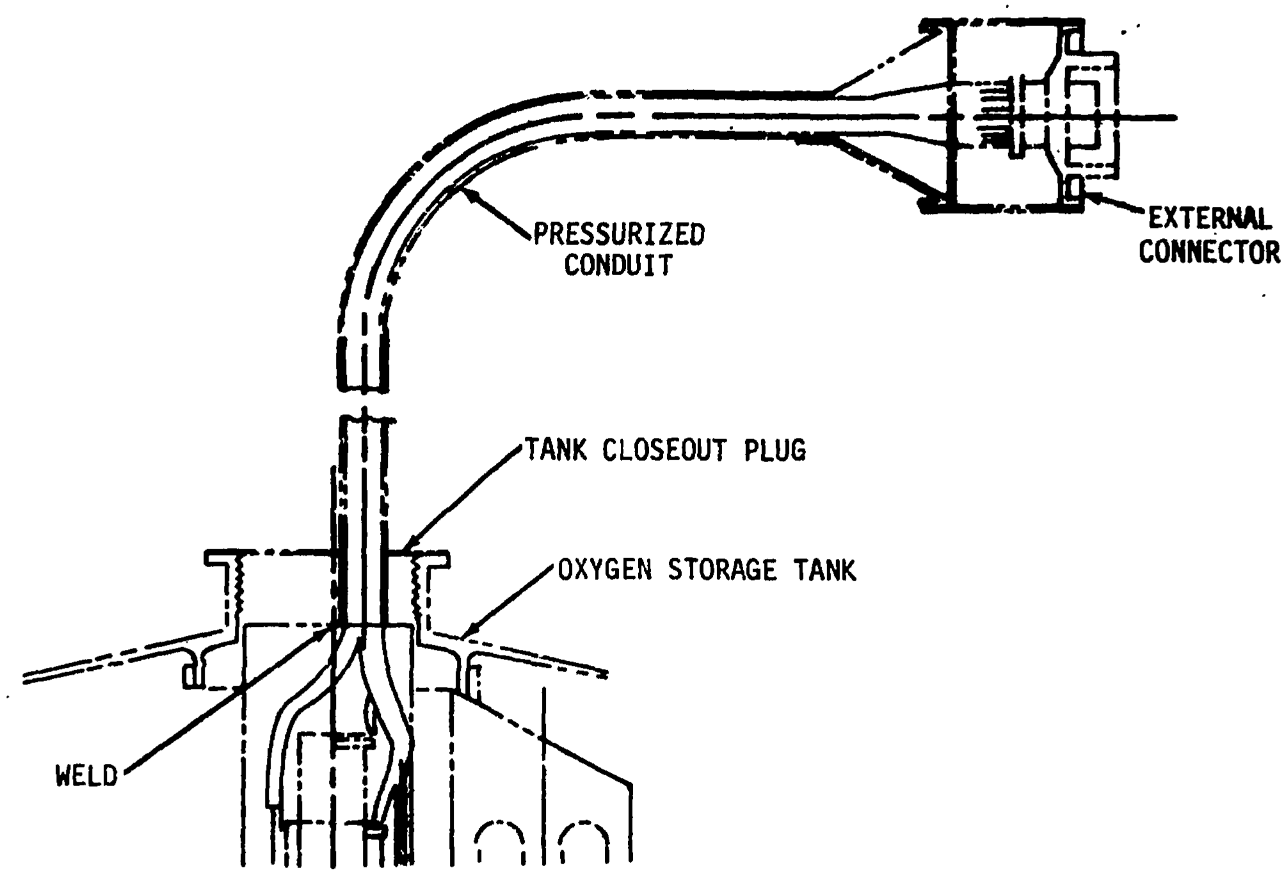


FIGURE 5. PROBE LEADS ROUTED THROUGH PRESSURIZED CONDUIT

Corrective action recommended by Panel 5A consists of modifying the spacecraft wiring which will provide 'barberpole' indication for either H_2 or O_2 reactant valve closure for each fuel cell, and tie this warning into the audible caution and warning signal. Because of this signal alert capability, the crew will have sufficient time to manually switch the affected valve to the open position.

4.3 ASSESSMENT OF CONFIGURATION DIFFERENCES AMONG CSM's 108, 109, AND 110.

Utilizing a detailed examination of the contractors indentured parts list, system drawings, and a computer printout of all CSM 108, CSM 109, and CSM 110 engineering order differences, it was concluded that no significant configuration differences existed between the vehicles surveyed. (Reference report - Appendix G). Based on this conclusion, no further consideration of configuration differences was made by the Panel.

4.4 FUEL CELL O_2 REACTANT SHUTOFF VALVE ELECTRICAL DESIGN.

The fuel cell reactant shutoff valve module contains two check valves and three isolation valves in one housing (Figure 6). The valve design is such that the wiring (lead in wires, valve coils, and position indicators) are exposed to the 900 psi oxygen. All wires, including the coil wires are coated with Teflon and the operating mechanism for the position indicators contains significant quantities of Kel-F. Also, since all three valves are in a single housing, a failure of one valve due to combustion of these materials may result in failure of the entire oxygen system or failure of all three fuel cells.

There are two basic approaches to alleviate these problems:

a. Modify the existing valve

b. Use some other Apollo valve

c. Use existing valves after extensive screening

The first alternative requires rework of the valve body to isolate the coils and the use of an externally mounted position indicator switch. These changes can be made without changing the basic configuration of the valve. The only functional change would be the switch and possibly some restructuring of the internal load sensitive members. Little or no spacecraft change would be required.

The second alternative is possible using the SM-RCS helium isolation valve and check valves. The SM RCS valves have all the features required, except the qualification for low temperature. The valves are presently qualified for -60°F and would have to be requalified down to -300°F. The existing valves were not cleaned for use in oxygen. Valves cleaned for use in oxygen would have to be procured. The majority of the problems associated with the use of these valves would be in packaging for the fuel cell application.

A simple tubing manifold can be arranged with the alternate valves to achieve a flow schematic identical to the existing configuration (figure 7a). However, tubing would be used instead of the single forged housing currently used. The other possible consideration is a complete redesign of the system wherein only a portion of the working fluid is lost in the event of a leak (figure 7b).

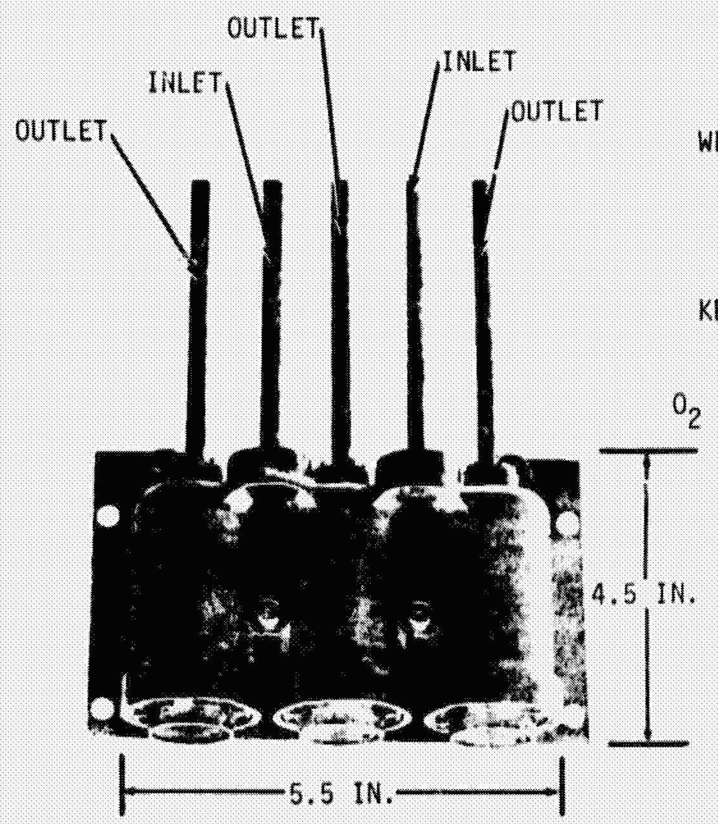
NR has been requested to evaluate these two approaches. The issues relative to the modified existing valve are whether a satisfactory redesign can be implemented, and whether a suitable valve position indicator switch can be incorporated. The issues relative to the alternate RCS valve are as follows:

a. The need to eliminate the possibility of single point failure.

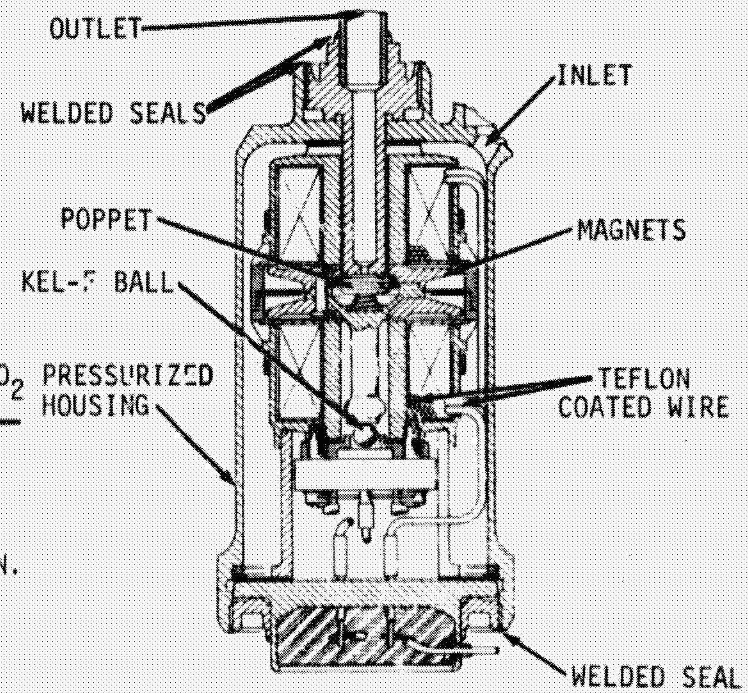
b. The extent of spacecraft wiring and switching modification required.

c. The adequacy of the valve for the intended service.

d. The availability of oxygen compatible valves.

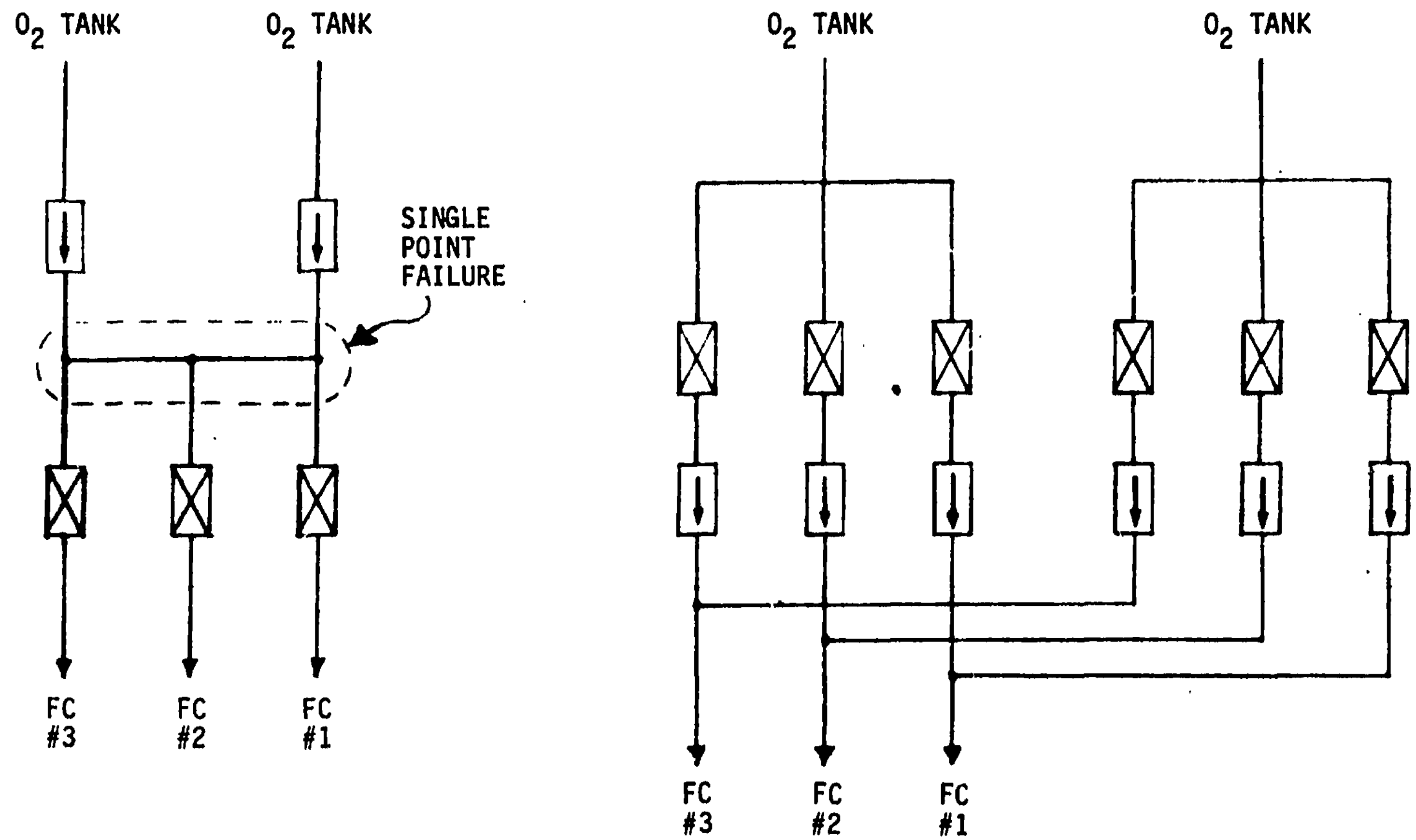


THREE LATCHING SOLENOID VALVES, PACKAGED IN A SINGLE HOUSING



CROSS-SECTIONAL VIEW OF APOLLO LATCHING SOLENOID VALVE (ACTUAL SIZE)

FIGURE 6. FUEL CELL REACTANT SUPPLY VALVE MODULE.



(a) ALTERNATE A

(b) ALTERNATE B

FIGURE 7. ALTERNATE VALVE CONFIGURATIONS.

APPENDICES

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APPENDIX A

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APPENDIX A

QUANTITY GAGING EVALUATION

An analysis was made for the purpose of determining capabilities of various methods of determining the inflight quantities of oxygen in the cryo tanks. The methods evaluated were as follows and are discussed in separate subsections of this report:

1. Pressure-temperature analysis
2. Heater cycle period
3. Total energy
4. Flow integration
5. Capacitance probe

1. Pressure-temperature (P-T) analysis. The specification accuracy of the temperature probe is ±2.68% of full scale or approximately ±10°F. The resulting inaccuracy in computing the quantity rises from ±5% with the tank full to ±25% with the tank 40% full. The mid range inaccuracy results from the thermodynamic characteristics of oxygen in the vicinity of the critical pressure-temperature point.

The known quantities, as determined by the existing capacitance probe and backed by consumption analysis, are consistently different than the quantities as determined by the P-T method. The P-T data is more consistent with the observed data and relatively linear if a -17°F bias is used throughout the quantity range. Apollo 12 quantity gaging data and P-T quantity determination both with and without the -17°F bias are shown in figure 1. There is no explanation for the existence of the bias. Figure 2 shows the error band (±10°F) associated with the temperature probe but based on the bias compensated nominal value.

2. Heater cycle rates. The system heater cycle rates as a function of tank quantity and flow rate are shown in figure 3. As can be seen in this figure, large changes in cycle rates are observed with relatively small changes in flow or quantity, especially in the region where P-T is most insensitive. The effect of the parameter during a mission is illustrated in figure 4. Various but constant flow rates are assumed, and the cycle rates shown for the ground elapsed times that correspond to the quantity that would occur. Also the Apollo 12 data is superimposed on the curve for correlation purposes. As can be seen, the flow rate compensates for the varying cycle rate and the curves for all the flows considered are essentially identical. No indication of a leak can be seen until the minimum specific heat input point (min dq/dm) is reached. As seen on figure 3, this occurs at a quantity of 37%. The cycle rates are not considered to be a useful tool for prime gaging purposes, but may offer some backup capability for confirming leakage where leakage initiates a finite time into the mission.

3. Total energy. This method requires an accurate determination of the quantity at lift-off or subsequent to servicing and an accurate determination of the total electrical energy applied to the tank from that point forward. The primary problem associated with this method is that small errors in determining the initial quantity are amplified later in flight. As an example, an error in quantity at lift-off will be amplified by a factor of over 4, at the point at which 30 percent of the oxygen actually remains in the tank. This phenomena results from the fact that higher energy is required to expel the oxygen at the high density range as compared to the mid range or at temperatures near the critical temperature. Another factor which seriously compromises the accuracy of this system is the tank to tank variation in the heat leak. For instance, if the heat leak was twice the nominal level the resultant electrical energy demand through 70 percent depletion would be reduced approximately 40 percent. Assuming the heat leak to be nominal would result in the interpretation of only 20 percent of the fluid depleted.

4. Flow integration. This gaging concept would require the accumulative summing of the three fuel cell flow meters and the one environmental control system flow meter to determine the total usage from the oxygen tanks. This system could not detect any leakage upstream of the flow meters. Also no flow meter location is possible where there could be no leak between the meter and the tank. An additional disadvantage is that flow meters will not indicate the relative usage from the two tanks.

5. Capacitance gage. The capacitance gage currently in use has a specification accuracy of $\pm 2.68\%$ of full range when used in conjunction with the fan-motors. The final location of the probe relative to the heater, and the degree of stratification present will determine the accuracy that can be obtained. It should be noted that the probe will read low if its sample is hotter than the bulk fluid and high if its sample is colder. Which of these will occur is not predictable. Figure 5 shows the observed changes of probe readings resulting from fan-motor cycles for Apollo 8, 9, 10, 11, and 12. The random nature of these samples is evident in this figure; however, some improvement is observed as the minimum dq/dm range is approached. This is to be expected since the effect of stratification seems to decrease as this region is approached and since this condition is influenced by the decreased thermal demand in this region.

Finally, it should be noted that the capacitance probe is presently used to provide loading information. This is a key consideration since all red lines are based to lift-off conditions.

CONCLUSIONS:

1. It is concluded that the capacitance gage is the only alternative suitable for the Apollo missions using two tanks.

2. If the capacitance gage is not used, a third tank will be needed to supply the additional margins for the H missions.

3. The effects of stratification on the gage accuracy in the current configuration is tolerable.

APOLLO 12 CRY O_2 QUANTITY

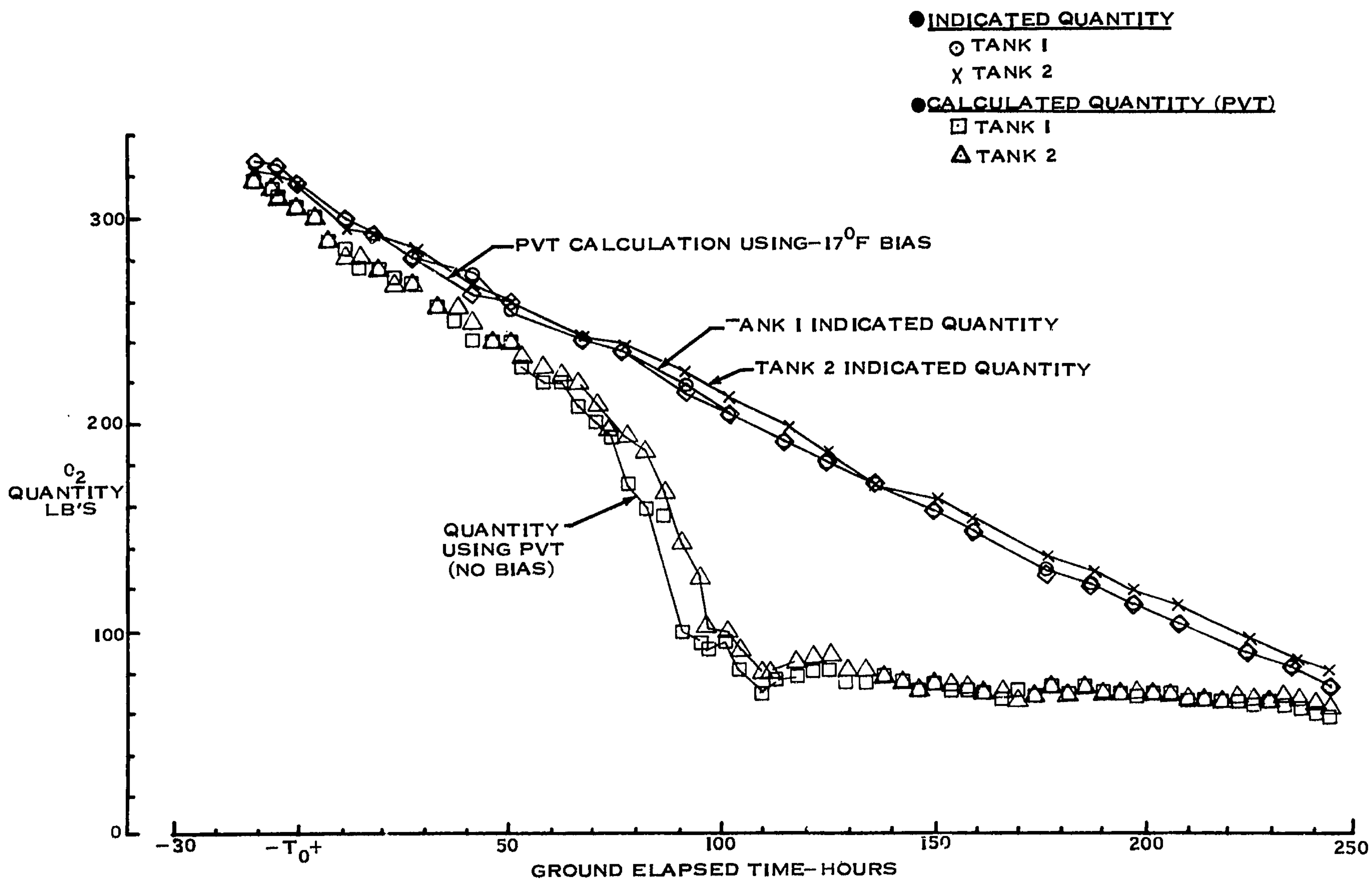

FIGURE 1

EFFECTS OF TEMPERATURE SENSOR ERROR

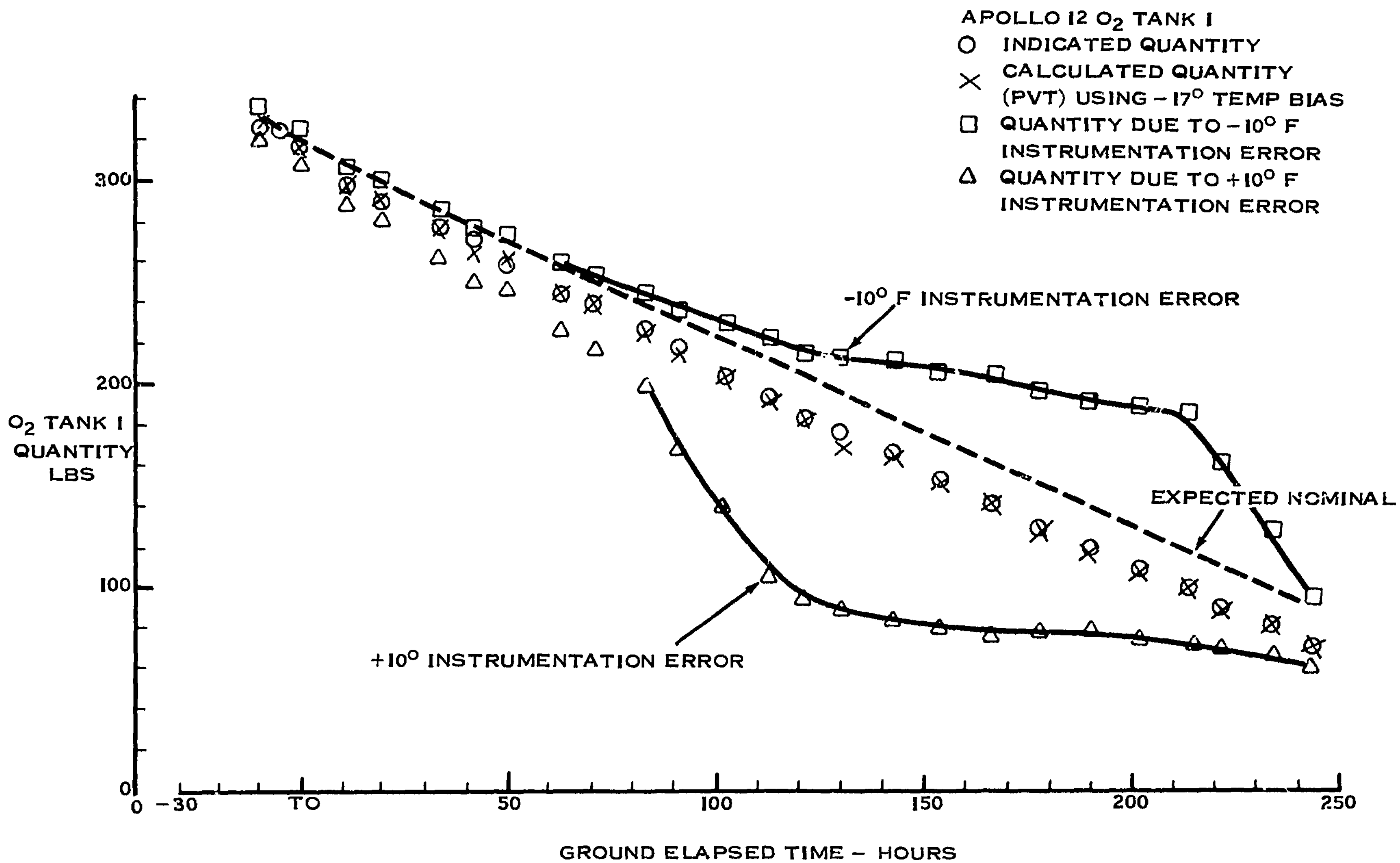

FIGURE 2

OXYGEN PRESSURE CYCLES SYSTEM OUTFLOW EFFECTS

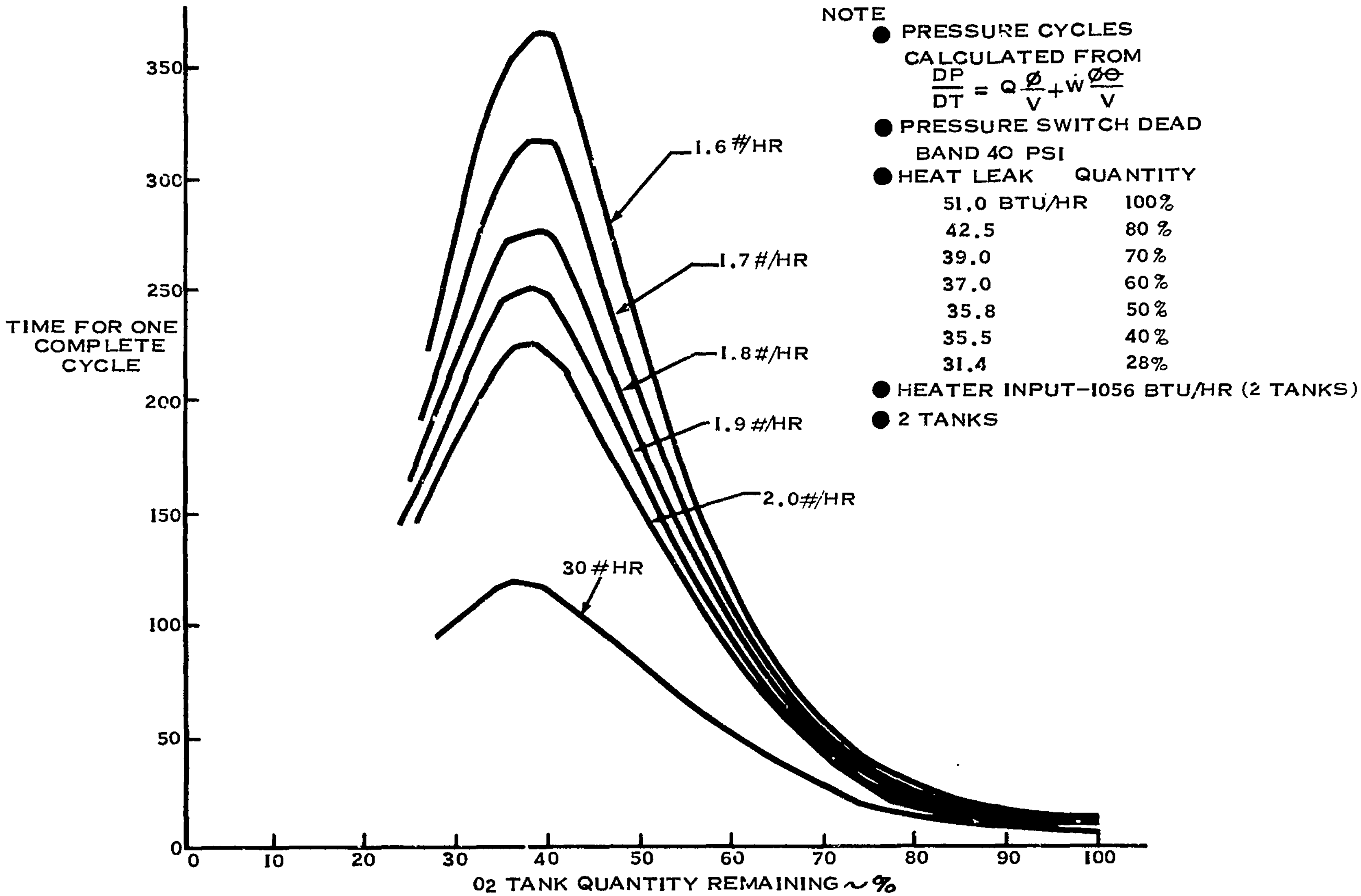

FIGURE 3

QUANTITY DETERMINATION USING HEATER CYCLES

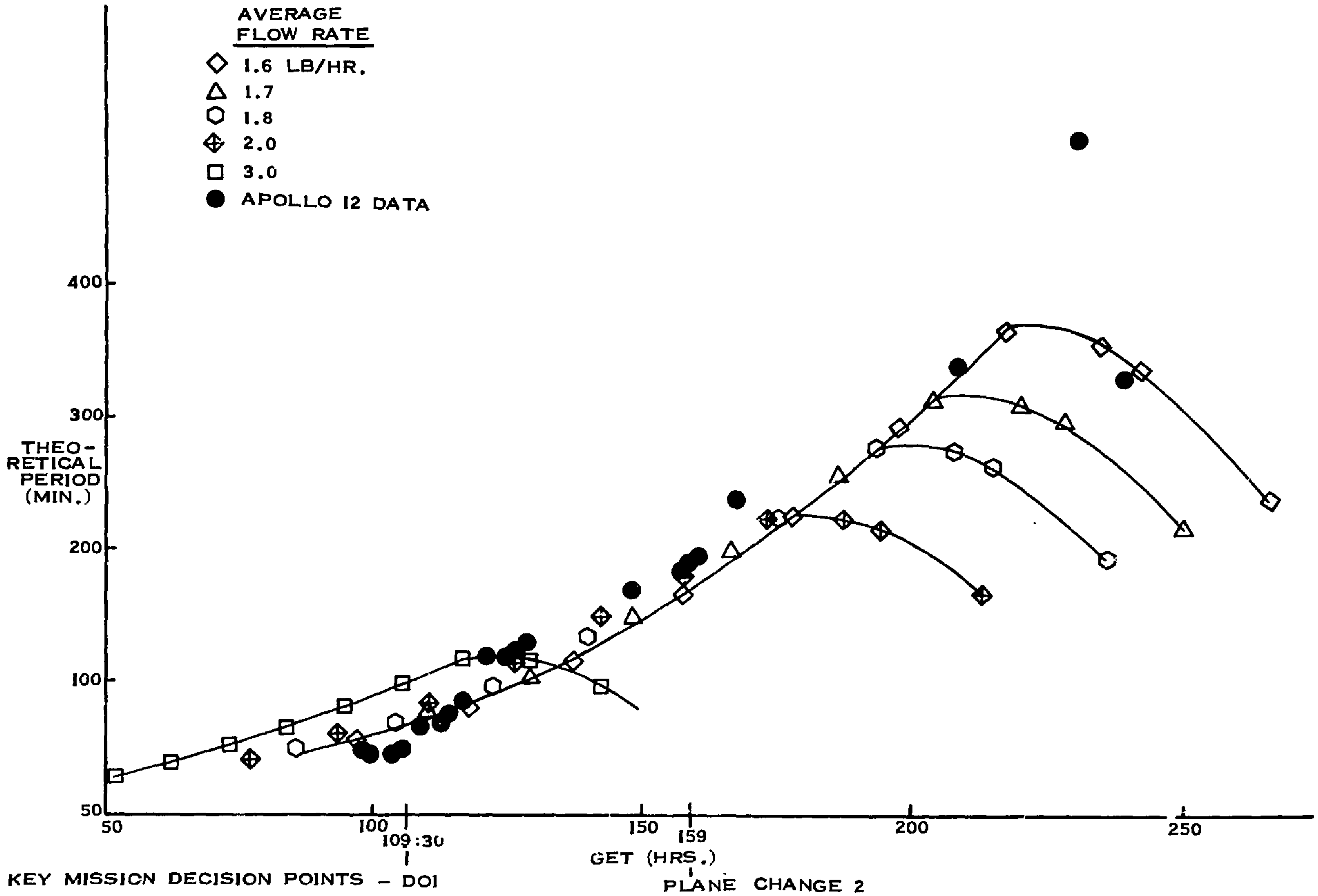

FIGURE 4

OXYGEN TANK QUANTITY CHANGE

CHANGE DUE TO STRATIFICATION

1% = 3.2 LBS. OF O_2

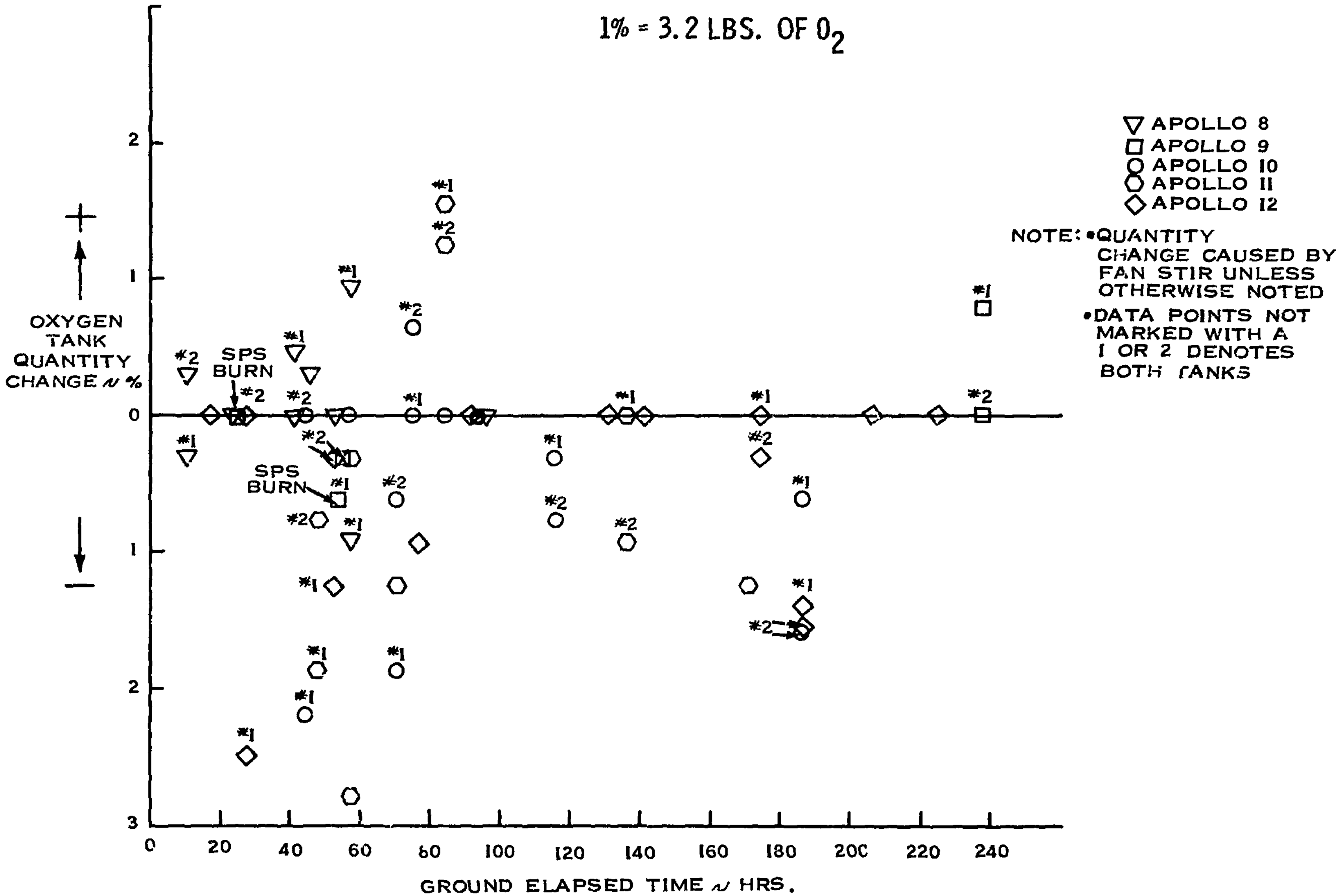

FIGURE 5

APPENDIX B

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
MAY 11 7 03 AM '70 HOUSTON, TEXAS 77058

IN REPLY REFER TO:

MEMORANDUM TO: PF/Panel 5.a. Leader

FROM : FC32/Panel 5.a. Member

SUBJECT : Methods of Oxygen Quantity Determination

The proposal that the oxygen quantity probe be removed from the oxygen tanks prompted an investigation to ascertain the quantity determination capability by alternate techniques. The alternate techniques considered are based on the following methods:

a. Using pressure and temperature in conjunction with tables of oxygen properties.

b. Using time between heater on cycles or using pressure decay rate.

c. Using a heat balance of the tank.

Discussion of the three methods follow:

a. Pressure/Temperature Method: Quantity variation with pressure within the nominal operating range of the O2 tanks is slight; therefore, the degree of success of this method rests solely on how well the temperature can be determined and how well it represents the average temperature of the fluid. Physical characteristics of oxygen operating in the pressure/temperature/density range that is employed by the cryogenic storage system produce a temperature/quantity relationship that is approximately linear for quantities above 70 percent and below 20 percent. In the intermediate range, the relationship is such that small changes in temperature are reflected by massive changes in density (thus quantity)(e.g., a change in temperature of 5 degrees will indicate 50 pounds change in quantity).

The temperature measuring device, being immersed in the fluid, should closely reflect the temperature; however, empirical data shows that it generates a signal, on the average, about 15 degrees higher than the temperature which would correlate with the observed pressures and quantities. Exactly what causes this is not certain but the temperature sensor must be mounted on something (i.e., the probe) and it has wires leading from it out of the tank. The mount and the wires provide a direct heat path from the tank exterior to the sensor. Whatever the cause, experience has shown that a bias must be determined and applied to the temperature to achieve correlation. The bias has been determined for the present tank internal configuration (See Figure 1); however, it appears very probable that the internal tank configuration will be significantly changed thus changing the temperature sensor environment and requiring a new bias determination which cannot be done without a quantity probe.

The temperature measuring device has an accuracy of $\pm$ 2.68 percent and a range of 400 degrees. The accuracy in degrees is $\pm$ 10 degrees.

With the fans removed the effect of stratification on the accuracy of the temperature sensor must be considered. The quantitative effect of stratification is not known and is difficult to determine because each occurrence is somewhat unique and not necessarily repeatable.

The maximum quantity variation which could have occurred on Apollo 12 due to temperature sensor tolerance alone was approximately 27 percent of full scale. (Figure 2 shows the variation possible due to temperature instrumentation error. Plotted are indicated quantities from Apollo 12 and the expected nominal. Apollo 12 oxygen usage was 0.16 lb/hr higher than could be explained. An average temperature bias was determined by using Figure 1. This average bias was reapplied to Apollo 12 pressure temperature and this data was plotted showing good correlation with the indicated quantity. Then the variation in quantity with +10° F and -10°F temperature instrumentation error was plotted. The other two lines show effect of two leak rates which could deplete the oxygen supply and not be readily detected. This number was arbitrarily reduced to 15 percent to cover a fairly wide operating region. An additional 10 percent was added to cover the uncertainty of the temperature bias and the uncertainty due to stratification with a total uncertainty of 25 percent, the Apollo mission could not have been flown with the electrical power configuration used (i.e., powered up all the way) and a 40 amp return capability on one tank. Hence, missions with only P/T gauging for O_2 would have to be limited to 9 days or less. It is not certain that this 25 percent accuracy can actually be attained, but it is certain that no better accuracy can be guaranteed.

b. Pressure Cycle Frequency/Pressure Decline Rate Method: There are several undesirable possibilities associated with this method but one particular aspect renders this method totally useless. As the tank quantities decrease toward the Min DQ/DM region, the period between heater cycles nominally become longer. A leak in the tank would cause more frequent heater cycles for a given quantity but would also cause the quantity to decrease faster which tends to decrease heater cycle frequency. For leak rates up to three pounds per hour, these effects oppose each other one-for-one until the quantity has decreased to 30 percent. The effect is total masking of a leak until it is too late to perform corrective action.

c. Heat Balance Method: This method requires integration of heat input into the tank. The uncertainty is cumulative. That is, the error begins with the loading uncertainty. All subsequent periods without data where heater cycles occur, add to the uncertainty. Variations in the heat leak (heat leak apparently varies during flight and a fixed number must be assumed for this method) further compounds the uncertainty. Heat leak variations could either give the appearance of a leak or mask a leak, and a failed heater element would also give the appearance of a leak.

d. These methods plus integration of a fuel cell and ECS oxygen flows are adequate for oxygen management but provide little leak protection against anything except the grossest leaks. They just will not provide sufficient quantity determination capability to allow mission durations of the length of Apollo 14, not to mention J-series missions, and yet, guarantee that the oxygen supply will not be depleted prior to entry interface. (See Figure 3 which shows Apollo 12 and 14 nominal O_2 quantity remaining profile with 25 percent quantity uncertainty.) It is therefore mandatory that a direct means of quantity determination be provided.

C. L. Dumis

cc:
FA/S. J. Sjoberg
FC/E. F. Kranz
FC3/A. D. Aldrich

FC3:CLDumis:elr

INDICATED TEMPERATURE VERSUS TEMPERATURE BIAS
APOLLO 12

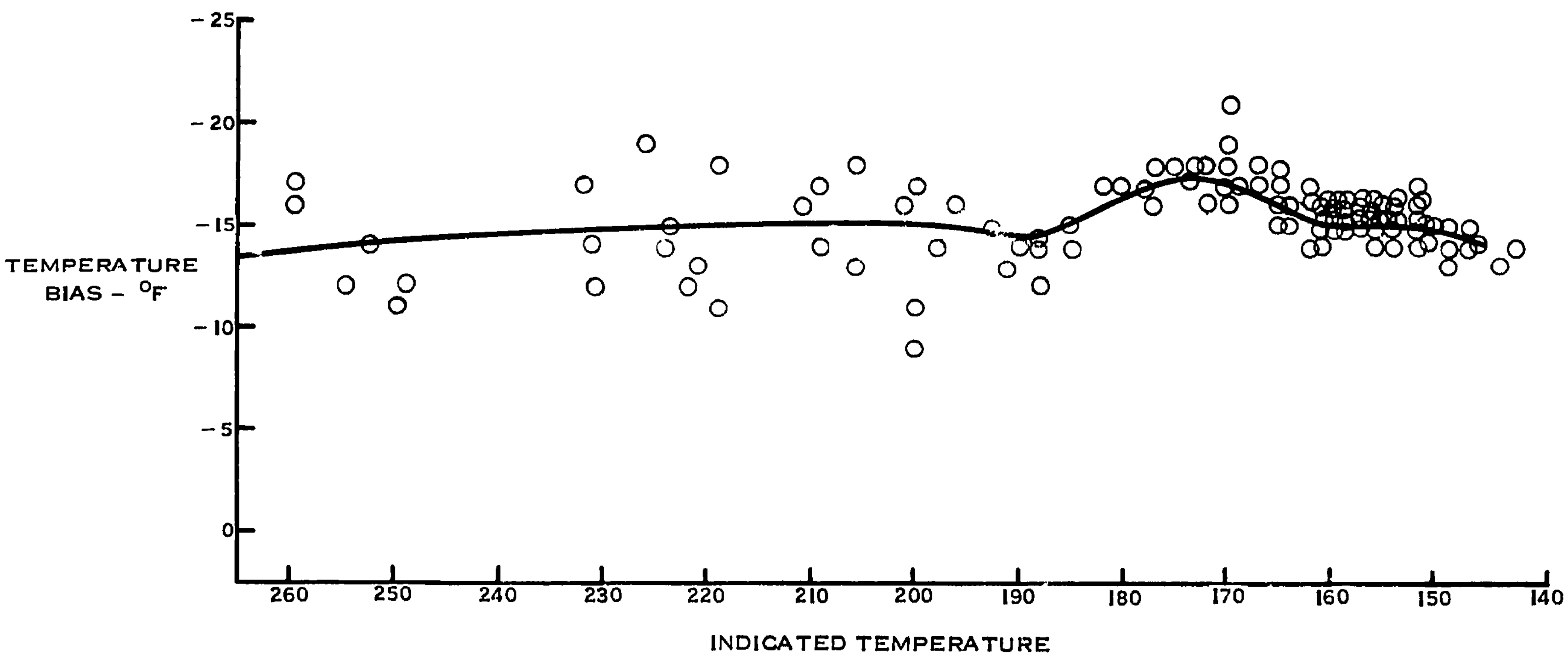

FIGURE I

APOLLO 12 O_2 TANK 1

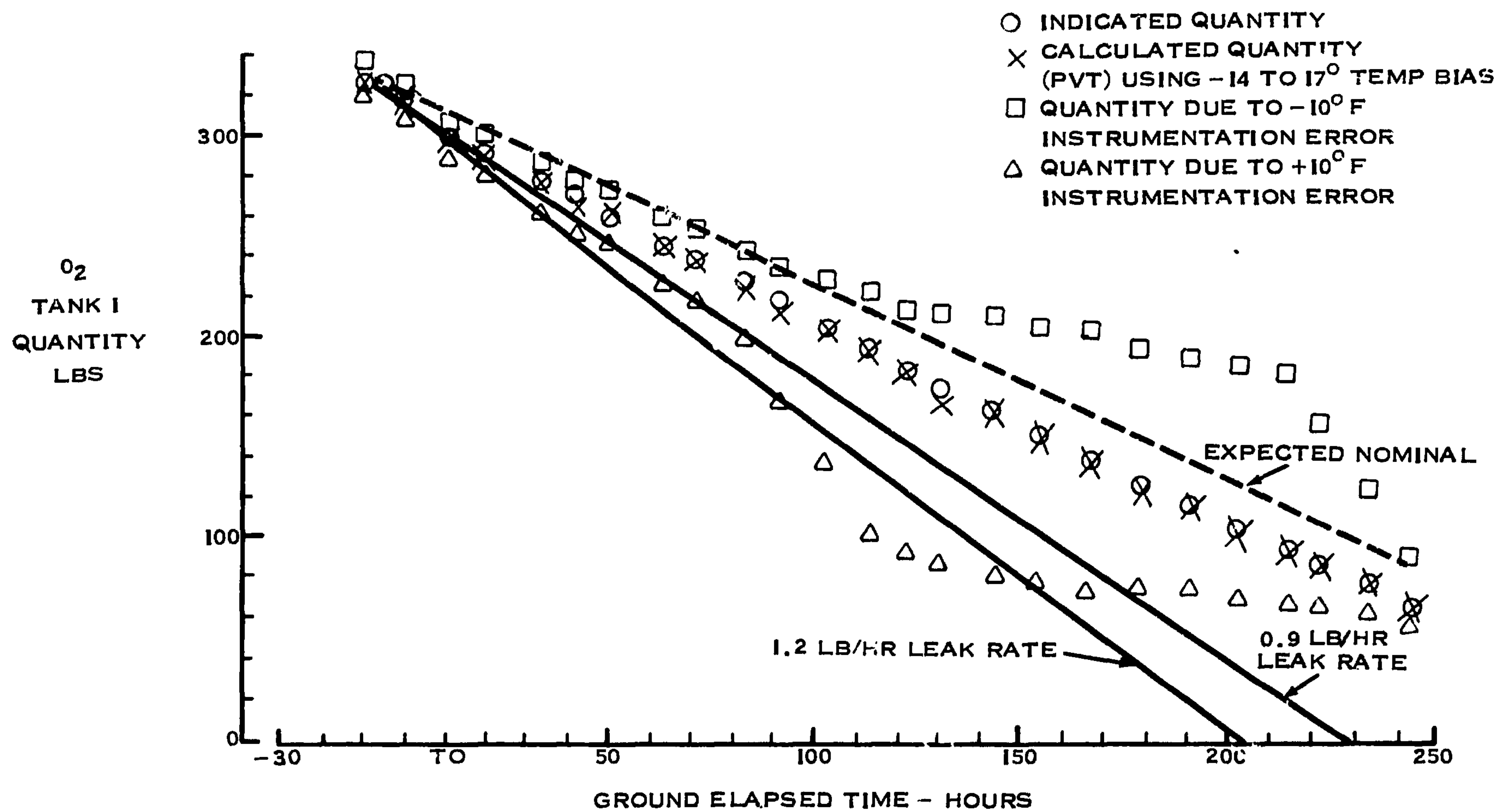

FIGURE 2

CRYO O_2 QUANTITY-LBS
VS
GROUND ELAPSED TIME

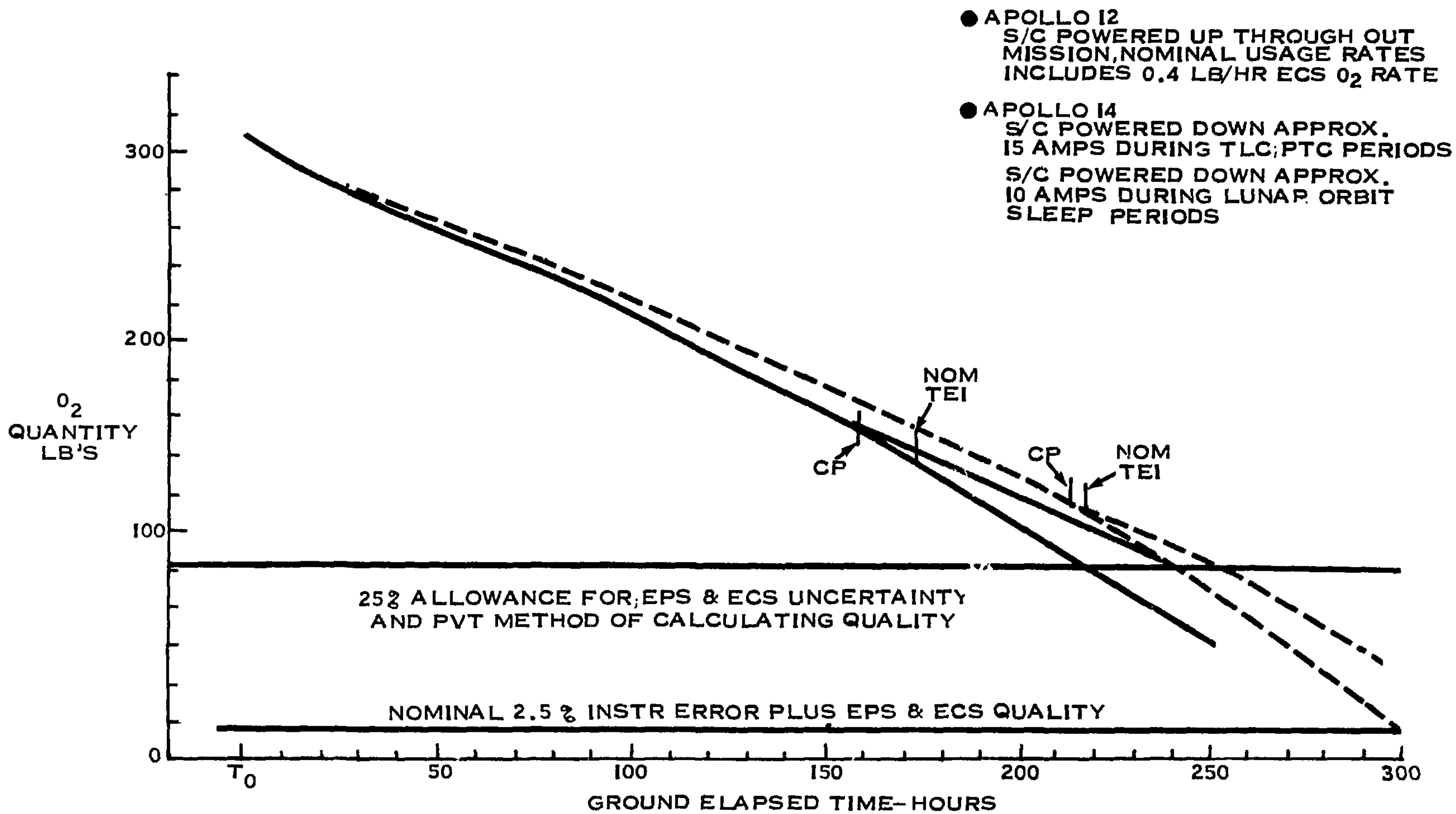

FIGURE 3

APPENDIX C

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Appendix C

STRATIFICATION ANALYSIS

Stratification is a phenomenon common to all fluids subjected to heat transfer. The density layers (strata) are caused by the temperature differences that must accompany heat transfer. In a one "g" environment these layers will segregate according to weight, but in a zero "g" environment they will remain static unless disturbed by some internal or external force.

If a high density supercritical fluid is stratified, and then mixed a pressure decay will normally result. This pressure decay can be explained by the following: heat energy is distributed heterogeneously to a portion of the fluid, and this heated portion, in turn, pressurizes the total contents of the vessels. The resulting pressure is not an equilibrium condition since an equilibrium condition is theoretically achieved by the homogeneous distribution of heat energy. If an unequilibrium condition is forced to an equilibrium condition by the redistribution of the total available energy a change in pressure can be expected because the fluid properties do not follow ideal gas laws in the area of concern.

The pressure decays that have been observed on the Apollo 7, 8, 9, 10, 11, and 12 flights are shown in figure 1. A drop of 145 psi on Apollo 12 is the highest drop that has been seen and all major drops have occurred before GET's of 100 hours. Figure 2 shows this same data superimposed on a pressure-quantity plot. The normal operating range, caution and warning limits and the minimum supply pressure are shown for reference. As can be seen all the drops occurred prior to the 60 percent quantity point. This does not necessarily imply that the temperature gradients are less but it does show that the effect on pressure control lessens as the critical temperature is approached and is probably influenced to a large extent by the reduced heat input in this region.

No flight data is available at low densities but the reversal in required heat inputs might be expected to reintroduce significant stratification. The mixing from this low density stratified condition should not have any significant effect on the pressure however since the fluid obeys the ideal gas laws in this region. A more significant effect in the low density region is the effect of stratification on the heater temperature. The low density (low quantity) fluid will not be as effective in dissipating heat from the heater unit as the colder high density fluid at the near filled condition. The problem of high flow rates from the system without the fan-motors has also been analyzed to some degree. The capability for supporting the flow rates in excess of 2.0 lb/hr or 1 lb/hr per tank without the fans has not been demonstrated in flight. The major concern that occurs due to these flows is that of heater temperature, and not the flow supporting capability or avoidance of two phase fluid conditions.

CONCLUSIONS:

1. The pressure drops caused by stratification and subsequent mixing can be tolerated.

2. The effect of stratification on the heater temperature is not known.

OXYGEN TANK
PRESSURE DROP DUE
TO STRATIFICATION

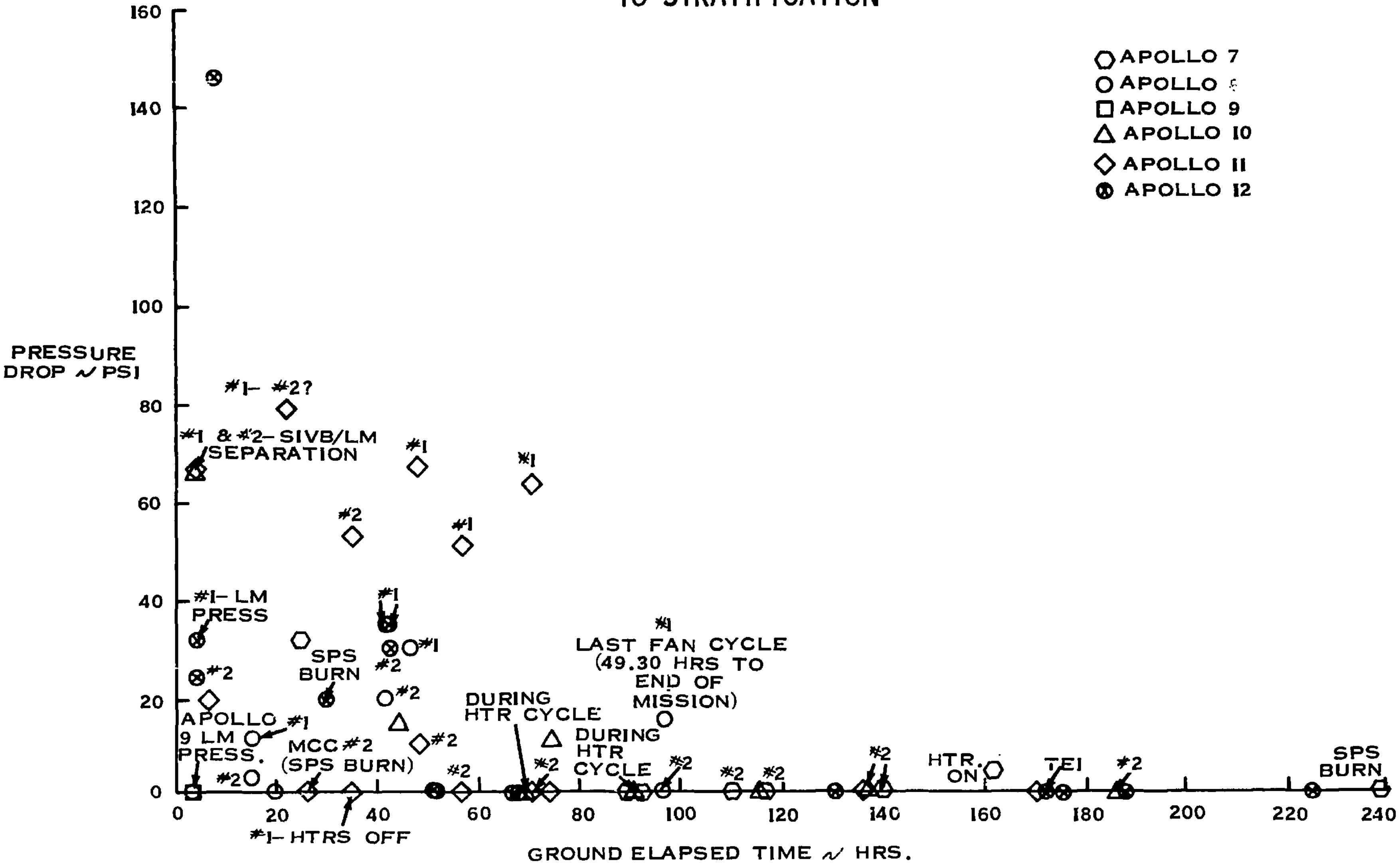

FIGURE 1

FLIGHT DATA PRESSURE DECAY
EFFECT SHOWN ON A
PRESSURE-QUANTITY DIAGRAM

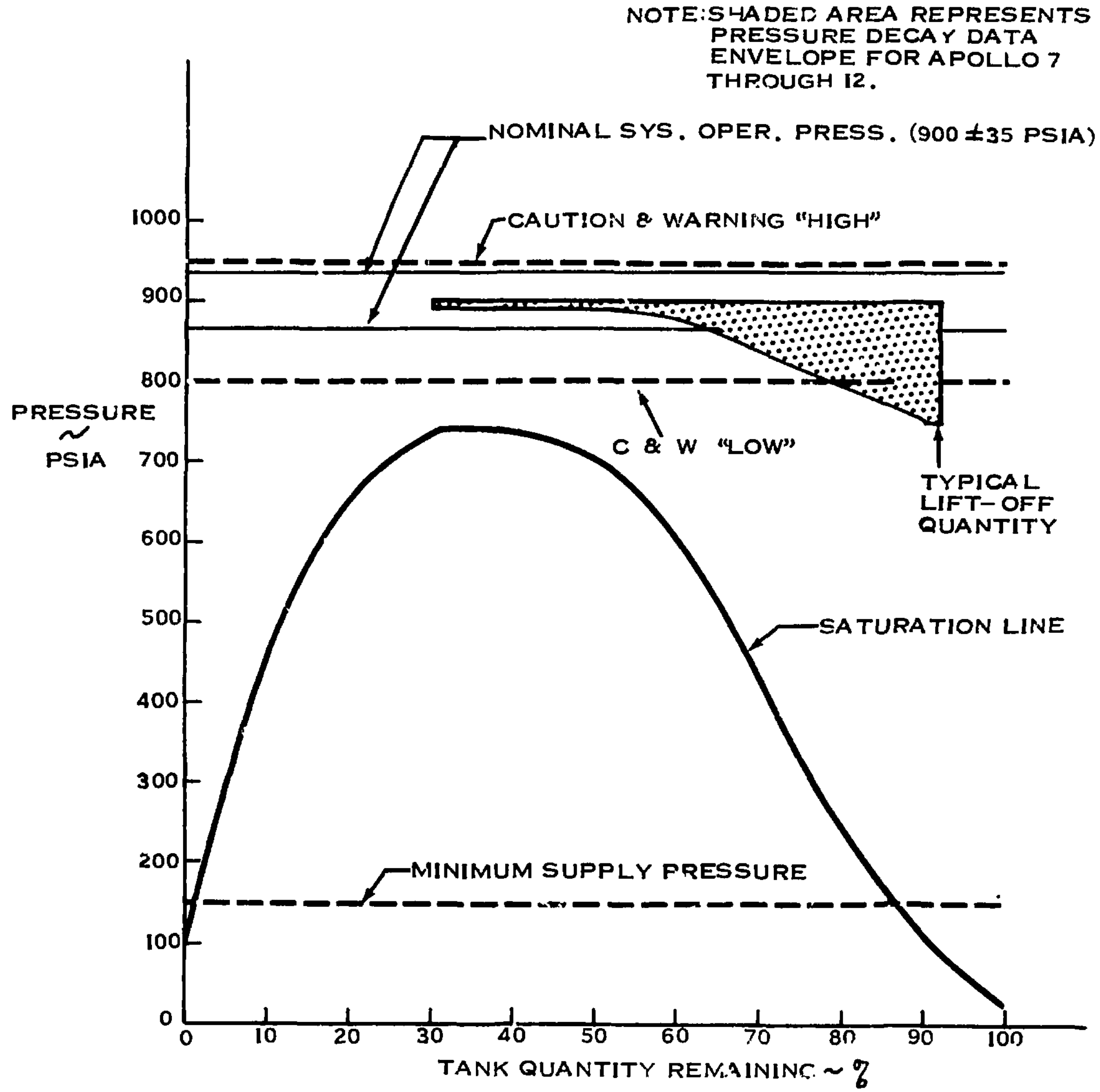

FIGURE 2

APPENDIX D

HEATER ANALYSIS

APPENDIX D

A. Heater Requirements - In conjunction with the redesign of the oxygen tank, a reassessment was made of flow requirements. Heater lead constraints, as discussed under design issues, limit the power dissipation per heater element to 42.5 watts. The existing two heater elements in each tank dissipate 57 watts each. Therefore, if the existing capability was a firm requirement, three elements would be needed in each tank. The flow capabilities of one, two, and three elements are shown in Figure 1. The effect of limiting the heater surfaces to +80°F during passive thermal control flight modes is also superimposed on the Figure. The actual limitation has not been determined, and the curve on the Figure is shown primarily to indicate that there is such a limit.

For the remaining two H missions, the nominal oxygen flow will be the same as experienced on previous flights or 1.6 to 2.0 pounds per hour. This flow is required at all times in the mission from either tank.

During the first twelve hours of the mission, a demand of 2.85 pounds per hour will continue to be a requirement for enrichment of the cabin atmosphere. Since this function is in preparation for continuation of the mission, one tank inoperative is not a consideration.

The LM repressurization following separation from the S-IVB also remains a requirement. The repressurization is accomplished from the gaseous oxygen tanks in the Command Module which are subsequently replenished from the cryogenic storage supply.

The H mission requirements are summarized in the table below:

	FLIGHT EVENTS		
	Nominal	O_2 Enrichment	LM Repress.
O_2 enrichment	----	0.70 lb/hr	----
Fuel cell demand	1.43 ± 0.12 lb/hr	1.65	1.65 lb/hr
Crew consumption	0.30	0.30	0.30
Cabin/suit leakage	0.20	0.20	0.20
Cabin/LM pressurization	----	----	<5.70
Total	1.93 ± 0.12 lb/hr	2.85 lb/hr	<7.85 lb/hr

SINGLE TANK
FLOW CAPABILITY

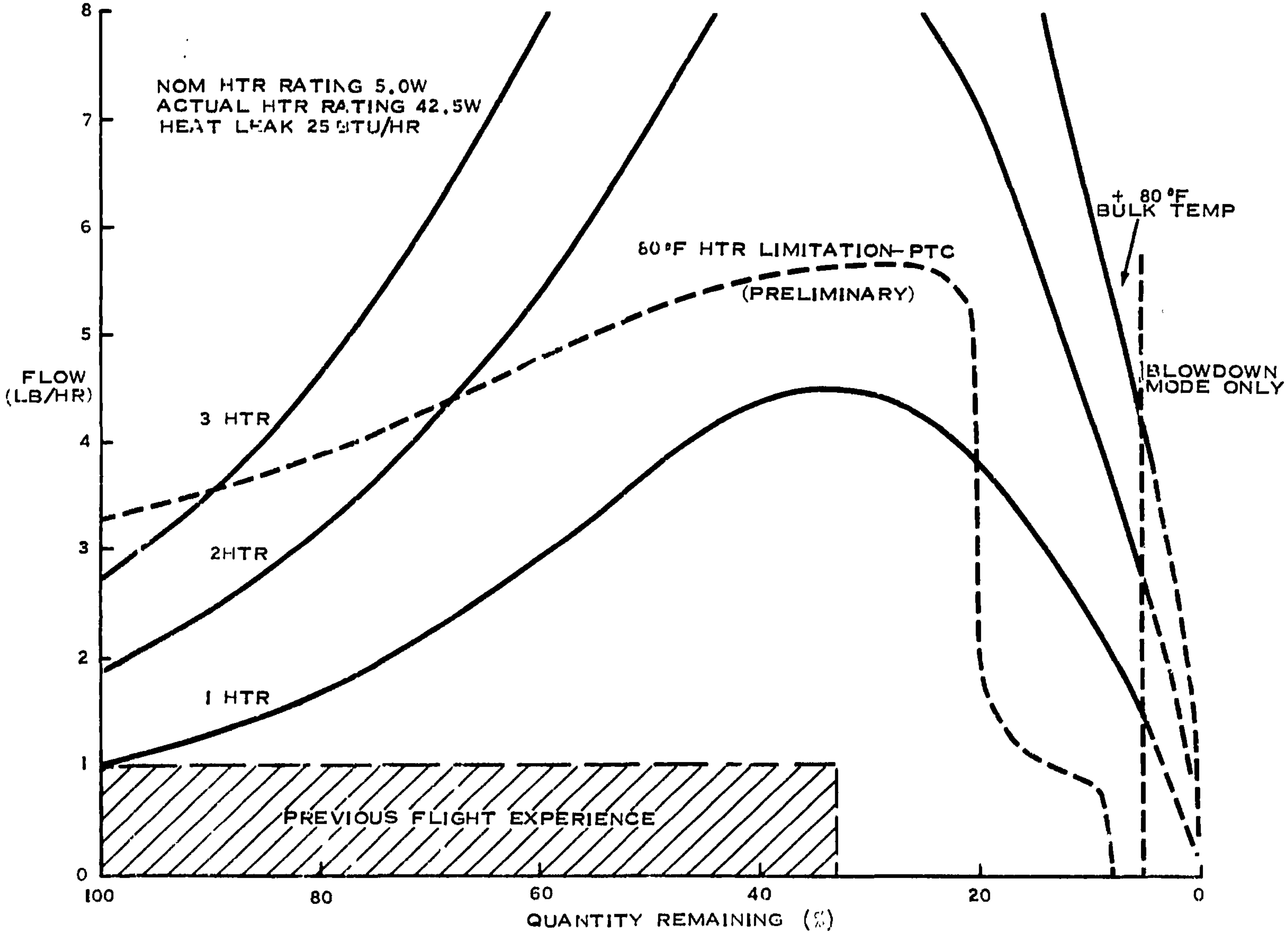

FIGURE 1.

J Mission

The nominal level of oxygen demand will be increased by that required to support approximately 500 watts of experimental load or approximately 0.37 pounds per hour increased fuel cell demand.

Additional requirements are associated with the EVA operations and resulting Command Module cabin repressurization. The maximum flow required during EVA will be 12.95 pounds per hour with a peak requirement of 8.49 pounds per hour during the subsequent CM repressurization.

The J mission requirements are summarized in the following table:

FLIGHT EVENTS

Function	Nominal	O_2 Enrich.	LM Repress.	EVA	CM Repress.
O_2 enrichment	----	0.70 lb/hr	----	----	----
Fuel cell demand	1.80±0.12 lb/hr	2.02	2.02 lb/hr	1.65 lb/hr	1.65 lb/hr
Crew consumption	0.30	0.30	0.30	0.20	0.30
EVA umbilical	----	----	----	11.00±1.00	----
Cabin/suit leakage	0.20	0.20	0.20	0.10	0.14
Cabin/LM press.	----	----	<5.70	----	6.40
Total	2.30±0.12 lb/hr	3.22 lb/hr	<8.22 lb/hr	12.95±1.00 lb/hr	8.49 lb/hr

B. Heater Capabilities - Nominal flows can be met with two heater elements regardless of whether one tank has two active elements or two tanks have one. Exception to this occurs early in the J mission where a third active heater is required. The other exception occurs when the tanks are nearly depleted. As shown in figure 2, the flow capability will probably be governed by heater surface area and not by the number of active elements in each tank.

The oxygen enrichment requires three active heaters and there is no restriction on whether this is the sum total of active heaters in one, two, or three tanks because the event occurs early in the mission where fluid temperatures are low. Mission rules would in all probability prohibit continuation of the mission if one tank was inoperative so a third heater element in one tank does not contribute to mission success in this region.

PRESSURE DECAYS FOR VARIOUS FLOW RATES AND HEATER POWERS
*ONE HOUR FLOW DURATION

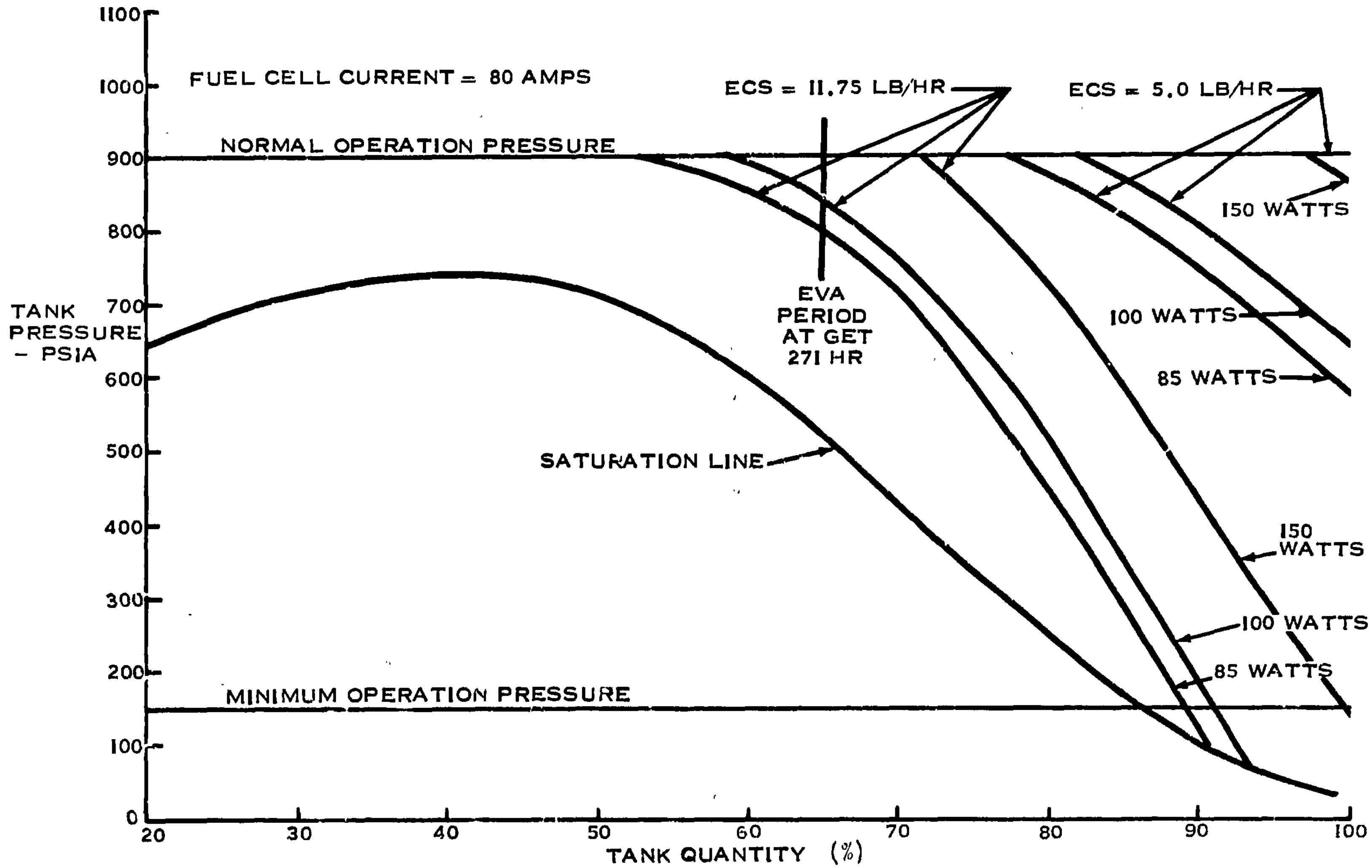

FIGURE 2.

H-Series Transients

An analysis was performed for high environmental control system (ECS) flow rates. Various flow rates were assumed for heater power of 35, 100, and 150 watts per tank. For the maximum possible flow to the ECS system (6.4 lb/hr) a decay of 487 psi was calculated at a quantity of 100% using two 85 watt heaters (i.e., 2 - 42.5 watt heater elements) in each of two tanks. This constitutes the maximum possible requirement for LM repressurization. The resultant pressure 422 psia is well above the minimum allowable (150 psia) and the fluid remains well into the single phase region as shown in figure 2.

J-Series Transients

For the worst case during an EVA a two tank configuration case was analyzed. Assuming the third tank which has the greatest flow capability at this point is lost. This flow as assumed to be 13.40 pounds per hour. Since the EVA is planned for a GET of 271 hours the minimum quantity possible in the two full tanks is 65% as shown in figure 2, this results in a maximum pressure drop of only 100 psia which is still 270 psi above the two phase region. Under the above extreme case only a slight further decay would be expected during the CM repressurization.

C. Heater Design Issues

Temperature Probe

There are no developed methods for determining the heater temperature under various modes of operation during a mission. The tanks flown to date have been equipped with thermal switches which are designed to open at +80°F. This level was set to coincide with the maximum allowable temperature of the pressure vessels. No indication of a thermal switch activation to the open position has been experienced in the flight programs to date. The flow demands, however, have been low as compared to the potential requirements and the densities experienced have not been less than those corresponding to the critical temperature. An area of specific concern is flow management below 20 percent quantity remaining where the bulk temperature rapidly increases and is +80°F with 5.5 percent remaining as shown in figure 3. The heat required per unit mass flow is also increasing rapidly in this region. A second area of concern is with high flow rates which have not been experienced in flights to date except for short duration and with the tanks essentially full.

A vigorous analyses is required of the heat transfer phenomena in the low gravity conditions as experienced in flight. Coupled with this is a requirement for managing the energy input in order to stay within heater temperature limitations. A reasonable approach would be the incorporation of a temperature sensor directly on the heater.

O_2 BULK FLUID TEMPERATURE
900 PSIA

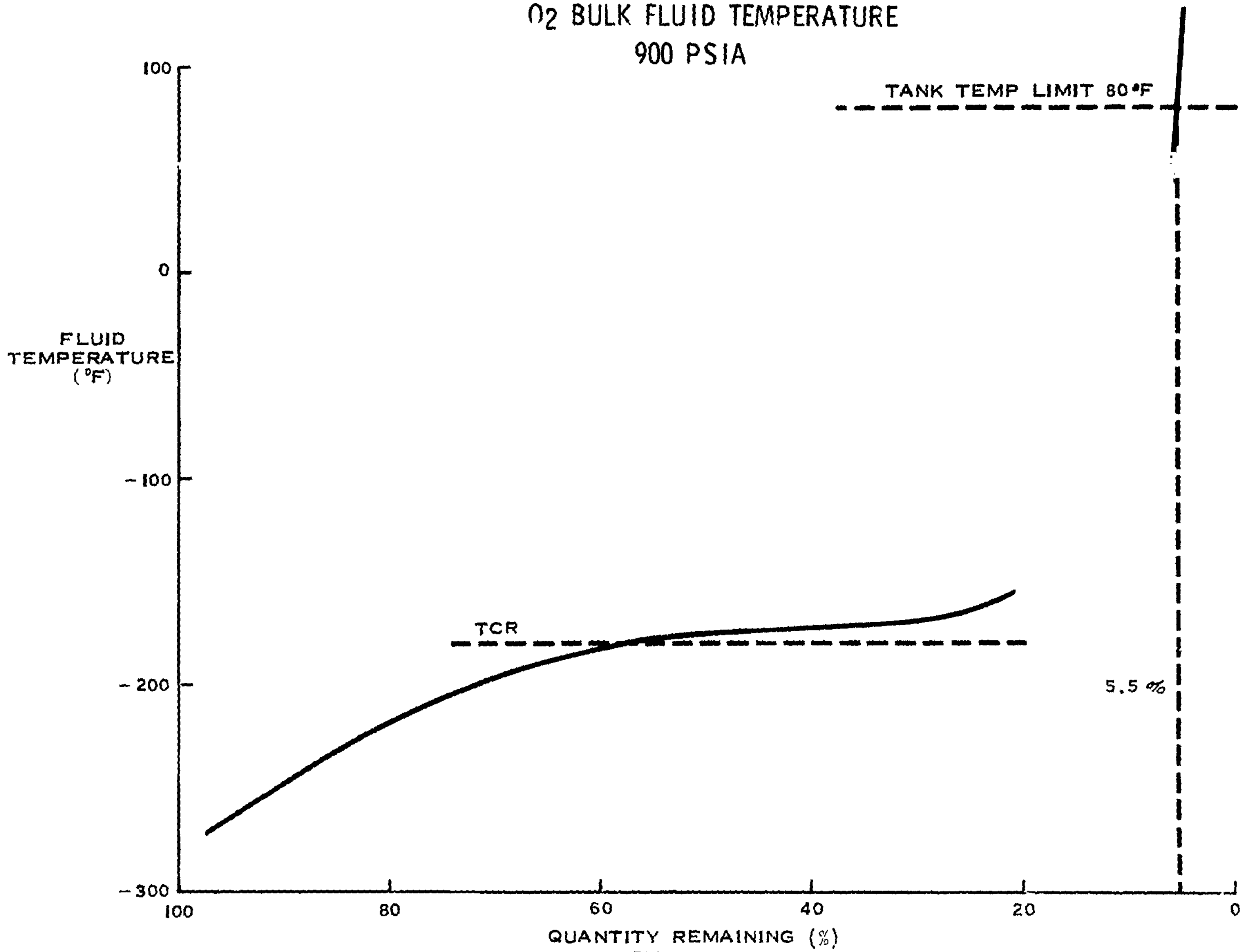

FIGURE 3.

Fin Effectiveness and Orientation

The tandem probe depends on extended surfaces of fins to achieve the equivalent heater area as the present heater assembly. Preliminary analysis indicates that the passive thermal control roll is a significant factor and that fins should be oriented to take maximum advantage of the flow pattern. With the heater probe in the X-axis of the spacecraft, the fins should be normal to the probe.

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APPENDIX E

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TEST PREPARATION SHEET

NASA - MANNED SPACECRAFT CENTER

1. TYPE		
A	Configuration Change	
B	Non-Configuration Change	

Mod. Sheet Number

2. TPS No. 13-T-43

3. S/C Cat. No.

5. Page 1 of 1

6. S/C No./Model No.: Apollo Blk I, O_2 Tank

7. Date: 5-8-70

8. Time:

9. Need Date: ASAP

10. Drawings, Documents, Ocp's, & Part Number(s): Ref: Apollo Fuel Cell & Cryo Support Handbook

OCP 0007

11. Contract Number:

12. Serial Number:

13. System: Apollo Blk I Oxygen Tank

14. Ref. E. O. Number:

15. TPS Short Title: Development of service Procedures for Apollo 14

16. Wt. Req.:

17. Reason for Work: Since the fan-motors have been deleted new procedures must be established to prevent severe stratification during the ground operations.

	18. DESCRIPTION (Print or Type)	21. Tech.	Insp. 22 CONT.	23 NASA
1	The KSC servicing procedures should be duplicated as closely as possible starting at approximately T-35 with LO_2 servicing.			
2	The one exception is that the system should be pressurized with 28 volts instead of with 70 volts.			
3	At T-0 (Launch) the system should be tumbled and the fan-motors run simultaneously.			
4	The GSE gear should be simulated for this test.			
5	Q.C. coverage will be required for this test.			
	Test will be performed after other cat. # 1 test are completed.			

EPG / C. Papp has Action

19. Prepared By: Robert R. Rice

20. Final Acceptance Date:

REFER TO PROCEDURES FOR REQUIRED SIGNATURES		REFER TO PROCEDURES FOR REQUIRED SIGNATURES	
Contractor	Date	NASA	Date
		[illegible signature]	[illegible] 70

MSC FORM 1225 (JUL 65)

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APPENDIX F

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MSC APOLLO 13 INVESTIGATION TEAM
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS 77058

REPLY TO
ATTN OF:

May 1, 1970

MEMORANDUM TO: A. Cohen/Leader, CSM Corrective Action Study & Implementation Panel (Panel 5.a)

FROM : R. Bobola/Configuration Assessment, Panel 5.a

SUBJECT : CSM 108, 109, & 110 Configuration Review Activity

The following summary is presented in response to the Apollo 13 investigation team leader's request to identify configuration differences among CSM's 108, 109, & 110.

A. A Systems Engineering Division (PD) review of the subject CSM fuel cell and cryogenic subsystems, using the NR identured parts list (IPL) and system drawings, revealed no design differences. The review did identify some drawing/accounting differences attributed to error, and known differences between certification hardware and flight hardware. (These are presented in detail in PD memo's PD5/M217, dated April 23, 1970, subject: Configuration differences in fuel cell and cryogenic subsystem; and PD5/M230, dated May 1, 1970, same subject - Enclosures 1 and 2.)

B. A NR review of a computer printout of all CSM 108, 109, & 110 engineering order (EO) differences, using the EO-accounting systems (EOAS) computer tape containing all Apollo hardware/software EO's released by NR, has identified some minor as-built configuration differences such as: an in line pressure transducer added to the CSM 109 hydrogen tank No. 2 as a replacement for the failed transducer internal to the tank and inaccessible, the deletion of a wire clamp bracket on CSM 109 fuel cell No. 2; the new service module jettison controller (SMJC) on CSM 109 and subsequent (25 sec. translation, 2 sec. roll); and transducer mounting/location variations among all CSM's. The nature of the EO differences substantiates the PD findings for fuel cell/cryogenic systems and holds true for all SM systems from a significant difference aspect. The NR review included CSM 111 and encompassed all SM system and component differences. Subcontractor support was utilized to assess component differences.

C. An expanded assessment of the complete Block II fuel cell/cryogenic systems configuration revealed that the only significant design change occurred on CSM 113 and subsequent. Thermoswitches, controlling cryogenic tank heaters, were replaced with straight heater circuits on CSM 113 and subsequent for H_2 tanks and CSM 114 and subsequent for O_2 tanks. The GSE fuel cell and cryogenic heater power supply was also modified to limit the duration of heater power application.

In summary, no design differences exist in the fuel cell and cryogenic systems for CSM's through 113; and no significant as-built total SM differences exist among CSM's 108, 109, 110, & 111.

R. E. Bobola

PF2:REBobola:drc 5-1-70

82

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC APOLLO 13 INVESTIGATION TEAM

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS 77058

REPLY TO ATTN OF: PD5/M217

APR 23 1970

MEMORANDUM TO: PF/Chief, CSM Project Engineering Division

FROM : PD5/Chief, Configuration Management Branch

SUBJECT : Configuration differences in fuel cell and cryogenic subsystem

Attached hereto is the first result of the configuration differences audit which you requested for CSM 108, 109, and 110 in the fuel cell, cryogenic and electrical power subsystem area. This audit is based on review of the configuration information on hand at MSC and will be expanded with completion of the review of the information received from NR.

Richard A. Colonna

Richard A. Colonna

Enclosure

cc:
Apollo 13 Investigation Team

PD5:RAC:dg 4-23-70

5-2731-HOU-134
April 21, 1970

To:	R. A. Colonna	MSC	PD5
cc:	D. G. Bradshaw	5-2700	HR-02
	D. H. Johnson	MSC	PD5
	R. E. Stevenson	5-2731	HA-69

Subject: Configuration Differences in the Fuel Cell and Cryogenic Subsystem

In conjunction with the MSC Apollo 13 Investigation Team Activities, we have searched the on hand MSC configuration records for the subject differences in Design Configuration for CSM 108, CSM 109, CSM 110 and the associated certification test units with the following conlcusions. We will check additional records as received from NR.

To date no design configuration differences have been identified in flight articles between CSM-108-109-110, in the fuel cell and cryogenic subsystem. The certification index for the vehicles, the differences display boards and configuration document have been reviewed and all lead to the same result.

Drawings have been ordered on the cryogenic subsystem and will be reviewed to assure that no differences exist between vehicles.

The review of CTR's indicates that some of the flight articles on the interested vehicles have been certified to Block I hardware, as a result, not all differences can be described between flight hardware and test articles.

The following is a list of only those items in the CTR where the flight article is different than the test article.

Harness Assembly Electrical Connector and Signal Conditioner

Flight article (ME148-0027-0010) is certified by similarity to Block I hardware. Flight hardware has a 24 pin in lieu of 19 pin interface and integrated solder cups instead of brazed solder cups; also 4 jumper wires were removed from connector so that a fan motor phase voltage check can be made. Improved shielding on the signal conditioner output.

Tank Assembly O2

Flight article (ME282-0046-0003) is certified by similarity to Block I hardware. Flight hardware has no Mylar bag (insulation) and additional connectors pins for fan motor verification. Also, vac-ion pump potting has been revised to eliminate corona effect, and an altitude test has been added to acceptance test.

Tank H2 (Including Harness Assembly)

Flight article (ME282-0047-0050) is certified by similarity to previous Block II hardware. Flight hardware has solder cup provisions for 24 wires instead of 20

Tank H_2 (Including Harness Assembly) Continued

to provide for individual fan motor phase wiring in lieu of parallel phase wiring. Also, vac-ion pump potting has been revised to eliminate corona effect and an altitude test has been added to acceptance test.

Valve Disconnect, Fill and Purge O_2

Flight article (ME284-0115-0001) is certified by similarity to Block I hardware. Data not on file at Boeing.

Valve, Vent O_2

Flight article (ME284-0119-0001) is certified by similarity to Block I hardware. Data not on file at Boeing.

Valve, Module, Shutoff, O_2 and H_2

Flight article (ME284-0289-0001) is certified by similarity to Block I hardware. Block II system uses identical valves for O_2 and H_2 system. Block I valves had different identification but no physical differences.

Fuel Cell

Flight article (ME464-0007-1002) is certified by similarity to Block I hardware. Data not on file at Boeing.

Valve Assembly Solenoid Operated

Flight article (ME284-0319-0001) is certified by similarity to Block I hardware. The flight article has the external mounting feet relocated to improve installation capability and changing the by-pass port to a male fitting.

Box Assembly Electrical Control, Cryo System

Flight article (V37-444020) is certified by similarity to previous Block II hardware, random acceptance vibration test has been added to the flight article.

Panel Assembly Cryo Control

Flight article (V37-445020) is certified by similarity to previous Block II hardware. Random acceptance vibration test has been added to the flight article.

The following items have been reviewed and the test articles are the same as flight article:

R. A. Colonna 5-2731-HOU-134

ME273-0041-0001	Coupling Assembly Disconnect, O_2
ME273-0047-0001	Coupling Assembly Disconnect, H_2
ME284-0290-0001	Valve, Module O_2
ME284-0291-0001	Valve, Module H_2
ME286-0036-0002	Filter-Inline
V37-447580	Cryo Fan Fuse Box

H. P. Wynn

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC APOLLO 13 INVESTIGATION TEAM

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS 77058

REPLY TO ATTN OF: PD5/M230

MAY 1 1970

MEMORANDUM TO: PF/Chief, CSM Project Engineering Division

FROM : PD5/Chief, Configuration Management Branch

SUBJECT : Configuration differences between CSM 108, 109, and 110

We have concluded the following review of the available documentation with respect to the radiator manifold installation (V37-45801), equipment installation cryogenic gas storage (V37-454200) and the fuel cell installation (V37-454001).

a. A drawing comparison has been made.

b. Drawings have been compared to the current indentured parts list.

c. Comparison of CTR test articles and the flight hardware has been made.

The following conclusions are drawn:

a. No design configuration differences exist between CSM 108, 109, and 110 in the above systems.

b. There are some differences in the accounting system and the drawings which are attributed to errors.

c. There are differences between some CTR articles and the flight hardware.

The drawing vs. IPL discrepancies and the differences between CTR articles and the flight hardware are defined in attachments 1 and 2. A letter to NR will be prepared to point out the drawing vs. IPL errors.

The serialized parts removal list for CSM 109 will be reviewed as soon as possible to verify removal and replacement of fuel cells at KSC with the latest configuration at which time a supplemental report will be issued if discrepancies are found.

Richard A. Colonna

Richard A. Colonna

Enclosure

PD5:TJJohnson:dg 4-29-70

cc:
PD/C. H. Perrine
PA/J. A. McDivitt
PA/S. H. Simpkinson

5-2731-HOU-134
April 21, 1970

To:	R. A. Colonna	MSC	PD5
cc:	D. G. Bradshaw	5-2700	HR-02
	D. H. Johnson	MSC	PD5
	R. E. Stevenson	5-2731	HA-69

Subject: Configuration Differences in the Fuel Cell and Cryogenic Subsystem

In conjunction with the MSC Apollo 13 Investigation Team Activities, we have searched the on hand MSC configuration records for the subject differences in Design Configuration for CSM 108, CSM 109, CSM 110 and the associated certification test units with the following conlcusions. We will check additional records as received from NR.

To date no design configuration differences have been identified in flight articles between CSM-108-109-110, in the fuel cell and cryogenic subsystem. The certification index for the vehicles, the differences display boards and configuration document have been reviewed and all lead to the same result.

Drawings have been ordered on the cryogenic subsystem and will be reviewed to assure that no differences exist between vehicles.

The review of CTR's indicates that some of the flight articles on the interested vehicles have been certified to Block I hardware, as a result, not all differences can be described between flight hardware and test articles.

The following is a list of only those items in the CTR where the flight article is different than the test article.

Harness Assembly Electrical Connector and Signal Conditioner

Flight article (ME148-0027-0010) is certified by similarity to Block I hardware. Flight hardware has a 24 pin in lieu of 19 pin interface and integrated solder cups instead of brazed solder cups; also 4 jumper wires were removed from connector so that a fan motor phase voltage check can be made. Improved shielding on the signal conditioner output.

Tank Assembly O_2

Flight article (ME282-0046-0003) is certified by similarity to Block I hardware. Flight hardware has no Mylar bag (insulation) and additional connectors pins for fan motor verification. Also, vac-ion pump potting has been revised to eliminate corona effect, and an altitude test has been added to acceptance test.

Tank H_2 (Including Harness Assembly)

Flight article (ME282-0047-0050) is certified by similarity to previous Block II hardware. Flight hardware has solder cup provisions for 24 wires instead of 20

Tank H2 (Including Harness Assembly) Continued

to provide for individual fan motor phase wiring in lieu of parallel phase wiring. Also, vac-ion pump potting has been revised to eliminate corona effect and an altitude test has been added to acceptance test.

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Valve Assembly Solenoid Operated

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The following items have been reviewed and the test articles are the same as flight article:

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ME284-0290-0001	Valve, Module O_2
ME284-0291-0001	Valve, Module H_2
ME285-0036-0002	Filter-Inline
V37-447580	Cryo Fan Fuse Box

N. P. Wynn

N. P. Wynn

5-2731-HOU-138
April 27, 1970

To:	R. A. Colonna	MSC	PD5
cc:	D. G. Bradshaw	5-2700	HR-02
	D. H. Johnson	MSC	PD5
	R. E. Stevenson	5-2731	HA-69

Subject: CSM-108, 109, 110 Fuel Cell and Cryogenic Configuration

Reference: Memo 5-2731-HOU-134 dated 4/21/70, N. P. Wynn to R. A. Colonna, Subject: Configuration Differences in Fuel Cell and Cryogenic Subsystem

The fuel cell and cryogenic subsystem is described by three major drawings; radiator manifold instl. (V37-458010), equipment instl. cryogenic gas storage (V37-454200) and the fuel cell instl. (V37-454001).

These drawings along with the indentured parts list (IPL) furnished by NR to NASA have been reviewed.

No design configuration differences have been identified in this system for the stated modules.

A discrepancy was noted in the fuel cell instl. drawing V37-454001 and the IPL's.

The drawing and IPL's indicate that fuel cells ME 464-0007-1000 are applicable for CSM-108 - 109, and that ME 464-0007-1002 is used on CSM-110 and subs. This is per ECP 11250 and RECP 9C124 in which the Block II secondary by-pass valves will be replaced with Block I valves. Although the fuel cell instl. drawing states that the -1002 fuel cell supersedes and is substitutable for -1000 fuel cell on CSM-108 - 109, it has not been updated to indicate that the -1002 fuel cells were installed in CSM-108 - 109.

The serialized parts removal list for CSM-108 indicates that the -1000 fuel cells S/N's 06763P650756, 06763P650757 and 06763P650758 were removed and replaced with -1002 fuel cells S/N's 06763P650752, 06763P650775 and 06763P650748 on 7/10/69 because of fuel cell #3 and its radiator loop was out of particulate specification limits after flushing.

The serialized parts removal list for CSM-109 is presently located at NR Downey therefore, the exact information is not readily available. I have been assured by our propulsion system engineers that the fuel cells on CSM-109 are -1002. This is in agreement with the historical Class I Differences Board which shows that the -1000 fuel cells were replaced with the -1002 at KSC.

In summary, even though a discrepancy was found in the drawing and IPL's for CSM-108 - 110 on fuel cell installations, no design configuration differences exist in this system for the stated modules. The serialized parts removal list for CSM-109 will be reviewed when it is available to this organization to verify removal and replacement of fuel cells at KSC with the latest configuration.

N. P. Wynn